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## EDITORIAL

Since the last issue of the *Journal* there have been several important developments in astronautics.

There was the launching of the first Russian *Sputnik*, which coincided with the assembly in Barcelona of the delegates to the Eighth International Astronautical Congress (reports will appear in the next issue of *Spaceflight*). This was a magnificent achievement and marked the opening of a new chapter in the history of science. The mass of the satellite was rather surprising, being about nine times that of the proposed Vanguard satellite, and thus indicating a take-off mass of some 80 to 100 tons. Even at that date it was evident that the Russian programme was much more ambitious than the American one, as well as being better expedited.

The first information obtained from the Russian experiment was that the decrease in atmospheric density with altitude was rather greater than had been expected; it seemed that *Sputnik* was to continue to circle the Earth for some months. After the "nine days' wonder," it faded from the front page headlines and in these circumstances even the success of Project Farside (in which an American balloon-launched rocket may have reached an altitude of over 4,000 miles) received little public notice.

However, both achievements are insignificant compared with that of *Sputnik* 2, with its canine passenger and the attempt at retrieval, and above all with the large mass involved (about a ton). One can only conclude that the Russians have a vehicle capable of landing an instrument package on the Moon, and then wonder what will be their next achievement. Perhaps a manned satellite?—or an unmanned circum-lunar trip? As this issue goes to press we await news of other satellite launchings—both Russian and American.

Is it not time that Britain took up the Russian challenge? No doubt our American friends will double and redouble their efforts, but are we to leave this important field of human endeavour to them and the Russians? If Britain on its own finds it too great an economic burden to undertake, could not it be a matter for Commonwealth co-operation?

# SOLID PROPELLENT ROCKET MOTORS*

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SUMMARY

The paper discusses the place that solid propellent motors hold in rocket propulsion, and describes various types of propellents and the ways in which charges can be designed. Some note is taken of erosion and instability, and the design of case components is described briefly.

## 1. Introduction

Solid propellents, no less than liquid propellents, are worthy of consideration by anyone contemplating the design of a rocket engine installation. There are fields in which the solid propellent rocket is clearly superior on the available performance, as in small air-to-air ordnance rockets, but there are also fields in which, although liquid propellent rockets may show a margin of superiority in performance, it may be preferable to use solid propellent rockets on grounds of ease of handling, simplicity, low first cost, and reliability. The feature, in liquid propellent rockets, of controllable thrust, is valuable, and it is not suggested that a solid propellent rocket is best for such an application as the propulsion of manned aircraft, but it is possible to build a programme of thrust variation into a solid propellent rocket for such duties as the launching of a satellite vehicle, for example.

It is the object of this paper to describe the way in which solid propellent rockets function, and to indicate the bases on which they are designed. Such rockets are simple in appearance, but nevertheless there are some quite complicated features of their design and performance. Fig. 1 gives an impression of a typical solid propellent rocket motor.

## 2. The Virtues and Vices of Solid and Liquid Propellents

Table I gives a brief comparison of some important features of typical solid and liquid propellents: the data given are not necessarily representative of recent practice. However, we may see that while solid propellents are fairly inert in storage, liquid propellents (especially oxidants) either evaporate or decompose: the only one in common use which has a fair storage life is red fuming nitric acid (R.F.N.A.) which is extremely toxic and corrosive. The lower limit of the storage temperature is defined, in the case of 80 per cent. hydrogen peroxide (H.T.P.), by the propellent freezing at — 24° C., and the fire risks with liquid propellents are quite alarming. The specific impulses of solid propellents are not as high as those of some liquid propellent combinations, and the cost of solid propellents (of fairly high specific impulse) is high: this is always likely to be the case, as most solid propellents have constituents which are either complicated in structure or which require the expenditure of large amounts of electrical energy (for example, ammonium perchlorate).

Such data as are given in Table I[1,2] indicate that solid propellents are to be preferred whenever it is likely that the auxiliary equipment for a comparable

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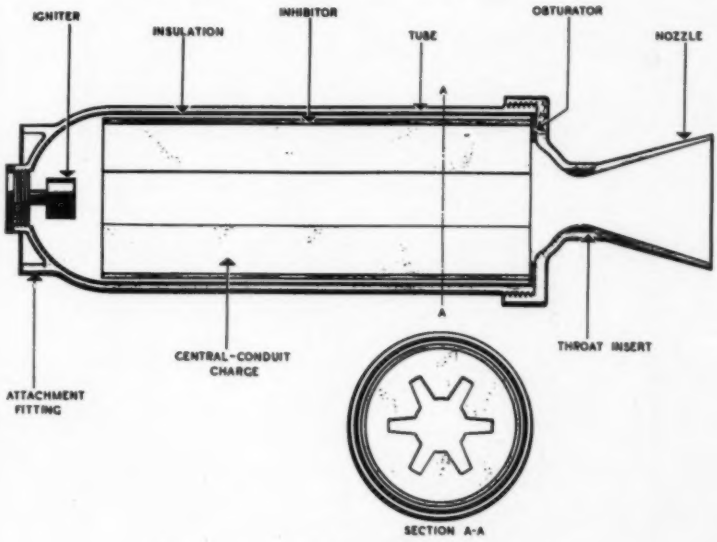


FIG. 1. Typical solid propellent rocket motor.

liquid propellent rocket may weigh an appreciable part of the weight of the propellents (as in small air-to-air missiles), where a rocket is required to be instantly available under almost all climatic conditions (as in sounding rockets for the investigation of showers of particles discharged from the Sun), or where it is not desirable or possible to have available a skilled team of men for care and maintenance at all times between manufacture and firing of the rocket (as in many weapon and sounding rocket applications).

Where sheer performance is the overriding criterion (as in ballistic missiles and satellite launchers), then it may appear that solid propellent rockets may not have much future. However, we must consider not only the specific impulse of the propellents alone but also the full and empty weights of the complete engine installation, for a given total impulse. Although solid propellent rockets, of necessity, submit the propellent container to the full burning pressure, the complete absence of pumps, pipes, turbines, auxiliary gas generators and control gear is a very favourable feature. The stresses in body cases are often almost entirely due to the pressure loads, the manœuvre loads being small by comparison, so that the body can be made of thin sheet metal without stiffening frames. In this way, solid propellent bodies can be made not much heavier than the tanks for liquid propellents, which often have to withstand some internal pressure to keep the pumps primed, and in addition they may have to be made of materials which are chosen for their compatibility with propellents rather than for their physical strength.

TABLE I

SOME FACTORS AFFECTING THE CHOICE OF PROPELLENTS[1,2]

| Property | Solid | | Liquid | | |
|---|---|---|---|---|---|
| | JPN (Double Base) | Resin/Ammonium Perchlorate (Composite) | Liquid Oxygen (LOX)/Gasoline | H.T.P./Kerosine | R.F.N.A./Aniline |
| Availability | Good | Good | Excellent | Excellent (Kerosine) Fair (H.T.P.) | Good |
| Cost | Dear | Could be cheap | Very cheap | Very cheap (Kerosine) Fair (H.T.P.) | Fair |
| S.I. (sec.) | 230 | 200 | 245 | 220 | 220 |
| Storage life | Indefinite | Indefinite | Continual evaporation (LOX) | Decomposes (H.T.P.) | Slight decomp. (R.F.N.A.) |
| Fire risk | Fairly high | Fairly high | High | Dangerous | Dangerous |
| Effect on organic material | Slight | None | Freezes (LOX) Dissolves (Gasoline) | Violent reaction (H.T.P.) | Reacts (R.F.N.A.) |
| Effect on metals | None | None | Embrittles (LOX) | Very corrosive (H.T.P.) | Very corrosive (R.F.N.A.) |
| Toxicity | Slight | None | Mild (Gasoline) | Poisons, burns (H.T.P.) | Very toxic fumes (R.F.N.A.) |

Having established a case for the use of solid propellents in rockets which may be considerably more effective than mere fireworks, we may now proceed to a discussion of some types of propellent and the way they burn and liberate the propellent gases.

## 3. Solid Propellents: Types and Behaviour

### (*a*) *Compositions*

The oldest and least useful solid propellent is gunpowder, a compacted mixture of charcoal, sulphur, and potassium nitrate. The specific impulse is so low (S.I. = 70 sec.) that we will not consider it further. A propellent which has been used for many years, in guns rather than in rockets, is cordite, which is a so-called double-base propellent. Briefly, this is a gelatinized mixture of nitrocellulose (guncotton) and nitroglycerine, with the addition of various stabilizers and plasticizers, which is made by a well established but comparatively complicated process. Cordite propellent is given its final shape by extruding it through a die fitted to a hydraulic press, and, since this process affects the consolidation of the propellent, it forms an essential part of the manufacture. The size and cost of large presses presents a limitation to the maximum diameter and weight of propellent that can be extruded in one piece. An alternative type of double-base propellent may be cast to shape in a mould: this removes the limitations on size and weight, but there are difficulties which have not yet allowed the extruded form to be supplanted.

Double-base propellents contain the oxygen required for combustion within the nitrocellulose and nitroglycerine molecules and form homogeneous gelatinized masses. A newer and different approach to propellent manufacture is typified by the composite propellents, which are made of mechanical mixtures of oxidants and fuels, held together with a resinous binder. Frequently, the oxidant takes the form of a ground-up inorganic salt, and the fuel may be provided by the binder itself, if present in sufficient quantity. Examples are the early asphalt-based propellents produced by Jet Propulsion Laboratory, and the more recent propellents based on ammonium perchlorate as the oxidant. Such propellents depend for their performance on intimate mixing of the fuel and oxidant and on proper control of the particle size, so that there is some likelihood of their being less reliable than double-base propellents as regards reproducibility of characteristics.

Composite propellents may be cast into rocket motor bodies as liquids and then cured into rubbery masses, by a suitable choice of binder. They are sufficiently pliable to allow themselves to be bonded to the inner wall of the motor, and yet to resist the formation of cracks produced by the stresses of thermal expansion and contraction. Double-base propellents, however, are not sufficiently flexible to permit this to be done, and a gap has to be left between the body wall and the charge to allow for differential expansion.

The specific impulses of solid propellents can be calculated from a knowledge of the calorific value of the constituents and of the mean molecular weight of the exhaust gases, or they may be measured. Properties of some solid propellents may be seen in Table II[1].

TABLE II

PROPERTIES OF SOLID PROPELLENTS

| Property | JPN | Alt—161 | Perchlorate |
|---|---|---|---|
| Type .. .. .. | Double Base | Composite | Composite |
| How made .. .. | Extruded | Cast | Cast |
| Specific gravity .. | 1·61 | 1·77 | 1·75 |
| Maximum storage temperature, °C. .. | 50 | 65 | 65 |
| Ignition temperature, °C. | 150 | — | — |
| Firing temperature limits, °C. .. .. | −30 to +60 | −55 to +75 | −55 to +75 |
| Lower pressure limit, lb./in.² .. .. | 500 | 1,000 | 200 |
| Smoke and toxicity .. | Nil | Smoky | Toxic |
| S.I., seconds .. .. | 230/1,000 lb./in.² | 186/2,000 lb./in.² | 200/1,000 lb./in.² |
| Burning rate, in./sec. at 20° C. .. .. | 0·65/1,000 lb./in.² | 1·6/2,000 lb./in.² | 0·5/1,000 lb./in.² |
| Pressure exponent .. | 0·69 | 0·7 | 0·4 |
| Temperature coefficient | 0·007 | 0·002 | 0·002 |
| Adiabatic flame temperature, °C. .. .. | 2,900 | 1,750 | 2,400 |
| Mean mol. wt. of exhaust gas .. .. | 28 | 30 | 25 |
| Ratio of specific heats | 1·21 | 1·27 | 1·26 |

### (*b*) *Behaviour during Burning*

The burning of solid propellents is understood to mean the reaction inside the motor body rather than the burning of a piece of propellent in air (if the propellent be fuel-rich there will be an additional release of heat due to combustion with the air), and one may make the following observations about such a reaction. When a piece of propellent is burning, its surface recedes in parallel layers: that is, flat surfaces stay flat, and circularly curved surfaces retain their centre of curvature as the propellent burns. The rate at which the surface recedes is a function of both the gas pressure above it and the temperature of the body of the propellent, and the steady burning is the result of heat being transferred from the reaction zone above the surface back into the solid. In the case of a homogeneous double-base propellent, the combustion process may be divided into three zones, described as the foam, fizz and flame zones.

These lie in sequence above the propellent surface: the foam reaction occurs in the solid surface itself, giving rise to gaseous products that react further in the fizz zone, which varies in depth inversely with the pressure: this accounts to some extent for the observed increase of burning rate as the pressure is increased. Above the fizz zone, the flame zone has a luminous reaction which releases all the remaining available heat and produces the final combustion gases.

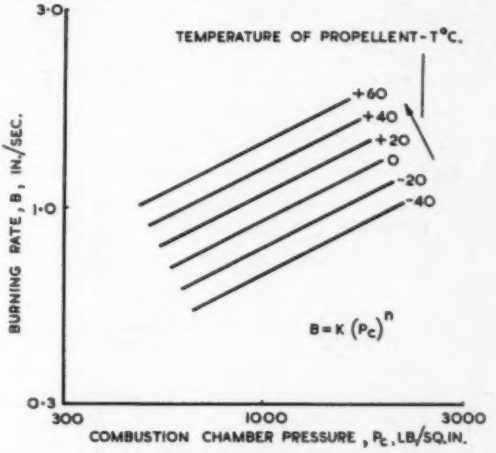


FIG. 2. Propellent burning characteristics.

The Allegany Ballistics Laboratory has proposed a theory for this behaviour, which indicates that the feed back of active radicals from the burning zones into the surface may control the burning rate. The behaviour of composite propellents is unfortunately not so well covered by theory. However, in general, for both types of propellent, we may say that[3] (see Fig. 2):

$$B = k(p_c)^n \qquad (1)$$

where:

$B$ is the rate of recession of the burning surface (in./sec.),

$p_c$ is the combustion chamber pressure ($10^3$ lb./in.$^2$),

$k$, $n$, are parameters: $k$ depends on the initial temperature of the propellent, and appears to be a function of the ignition temperature of the propellent, and $n$ is the burning rate/pressure exponent, which should be as small as possible; it is typically between 0·4 and 0·8. With the units given above, $k$ has values between 0·001 and 0·05.

The parameter $k$ depends on the propellent temperature, and it is usual to employ a temperature coefficient of the form:

$$\left(\frac{\delta \log p_c}{\delta T}\right)$$

taken at a constant ratio between the burning surface area of the propellent and the nozzle throat area. This may have values, typically, from zero up to 0·005 per degree Celsius.

The effect on burning performance of the exponent $n$ is very great: it is easy to show that, for a steadily burning rocket, the equilibrium burning pressure $p_{eq}$ is given by:

$$p_{eq} = \left[\frac{A_p k(\rho_p - \rho_g)}{A_t C_D}\right]^{\frac{1}{1-n}} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (2)$$

where:

$A_p$ is the burning surface area of the propellent,

$A_t$ is the area of the nozzle throat,

$\rho_p$, $\rho_g$ are the densities of the propellent and the gas above it,

$C_D$ is not a discharge coefficient in the usual sense as in hydraulics, but is defined by the relation:

$C_D \cdot p_c \cdot A_t =$ mass flow rate through nozzle.

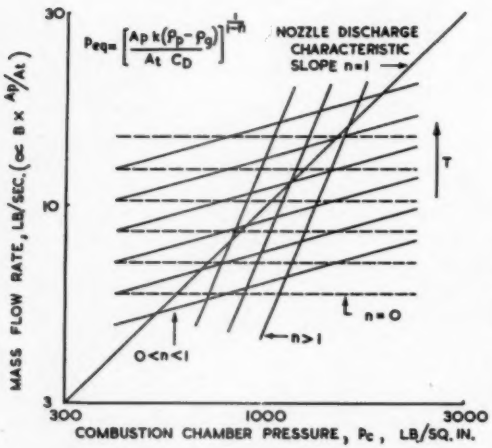


FIG. 3. Matching of nozzle and propellent.

We see in Fig. 3 that if $n$ tends to unity, the equilibrium pressure varies greatly for small changes in $A_p$ and $k$ (which are the most likely parameters to vary), while if $n$ is zero the dependence of $p_c$ on the bracketed variable in equation (2) becomes linear. If $n$ be greater than unity, the motor cannot burn stably at all.

Besides the dependence of burning rate on the gas pressure and propellent temperature, there is yet another cause for variation, namely the velocity of the combustion products in a direction parallel to the propellent surface: to a good approximation, this "erosive" burning rate is given[4] (see Fig. 4) by:

$$B = B_0(1 + k_v V) \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (3)$$

where $B_0$ is the burning rate with zero tangential gas velocity,

$k_v$ is a constant for the propellent,

$V$ is the tangential gas velocity.

It is apparent that if we have a perforated (tubular) charge, in which the gas flows out of the conduit at the rear end, the erosion of the charge will be greatest at the rear, and it will increase as the nozzle throat area is made larger and approaches the initial cross-sectional area of the conduit.

All these variations must be considered during the design of the charge of a solid propellent rocket motor.

## 4. The Choice of a Charge Shape

So far, we have considered only the behaviour of a piece of propellent in a particular environment of constant temperature, pressure, and gas flow velocity. However, if we have a lump of propellent burning in a vessel fitted with a nozzle of given size, the instantaneous burning rate, and the pressure, will depend on the instantaneous value of the ratio between the areas of the burning surface of the propellent and of the nozzle throat, as we have seen from equation (2). Since any rocket body must be designed to withstand the maximum pressure applied to it, even though the time of application be very small indeed, it is obviously wasteful to use a relatively heavy body when most of the burning will take place at low pressures, apart from a short period of peak pressure. In addition, it is usually desirable that the thrust should not exhibit peak values which would give objectionably large accelerations to the vehicle. Most solid propellent rockets, therefore, are designed to give a thrust /time curve of an almost rectangular shape, although the thrust may be required to rise or fall to some extent during burning to suit aerodynamic or other considerations (Fig. 5). The problem of charge design is thus one of shaping the propellent

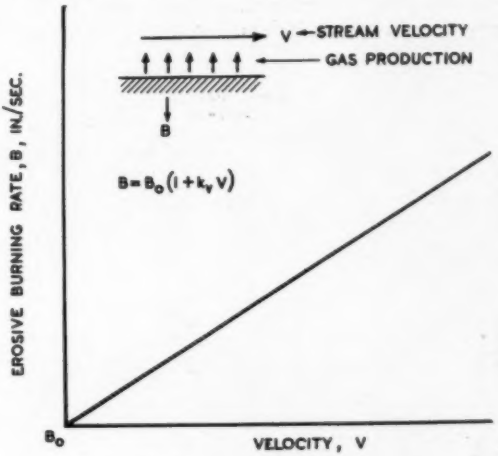


FIG. 4. Erosive burning.

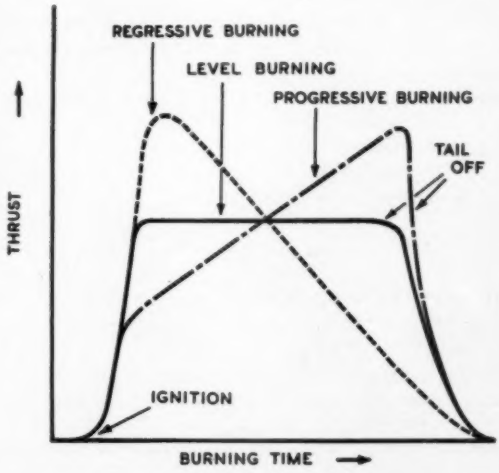


FIG. 5. Thrust-time curve.

charge so that, as the burning surface recedes, the exposed area of propellent remains substantially constant, or varies in a known and prearranged way. The experimental observation already mentioned, that the propellent surface always recedes in a direction normal to itself, has permitted a theoretical approach to be made to optimum charge design,[5] but it will have to suffice here to describe some of the typical charge shapes which have proved successful for different applications. In the following discussion, we must remember that some surfaces of the charge may be restricted, or inhibited from burning, by coating them with some non-reactive material such as cellulose acetate or ethyl cellulose: in the case of composite propellents, the charge may be inhibited by virtue of its adhesion to the inner wall of the motor tube. For example, the mode of burning of a stick of cordite may be changed completely by the way in which it is inhibited: if only the flat ends are coated, the propellent will burn radially inwards, releasing the gas in a short time, but if it be wrapped in a tube of inhibition it will burn on the ends only, like a cigarette, and last a long time.

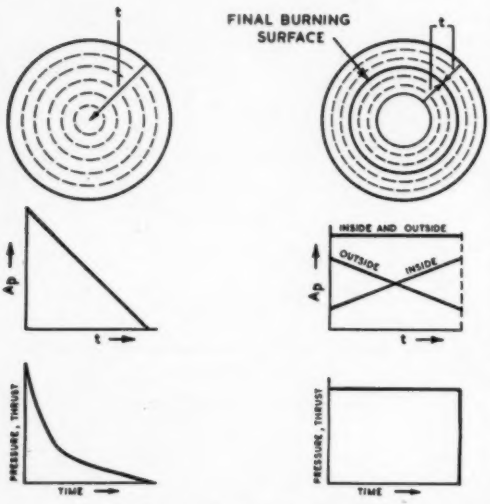


FIG. 6. Uninhibited charges.

Considering uninhibited charges first, it is apparent that since hot gas is evolved all over the surface of the charge, the rocket body will become heated by the hot gas flowing over it, usually to an excessive temperature, unless some insulating material is applied to the inner surface. This insulation will be heavy and will take up valuable space, so that uninhibited charges are not so suitable for high-performance rockets as are inhibited ones. It is, however, useful to consider the burning characteristics of two uninhibited charges (Fig. 6). A long solid stick of propellent, burning on the curved surface, has the surface area diminish as the diameter decreases during burning: thus the pressure/time curve falls rapidly. A tubular charge, burning on the inside as well as the outside, however, maintains the sum of the two surface areas constant, and so the pressure/time curve should have a flat top.

Turning now to inhibited charges (Fig. 7), the simplest of these is a circular tube with an external coating: the motor tube wall is protected, but since the surface area increases as the charge burns outwards, there is a sharp increase of

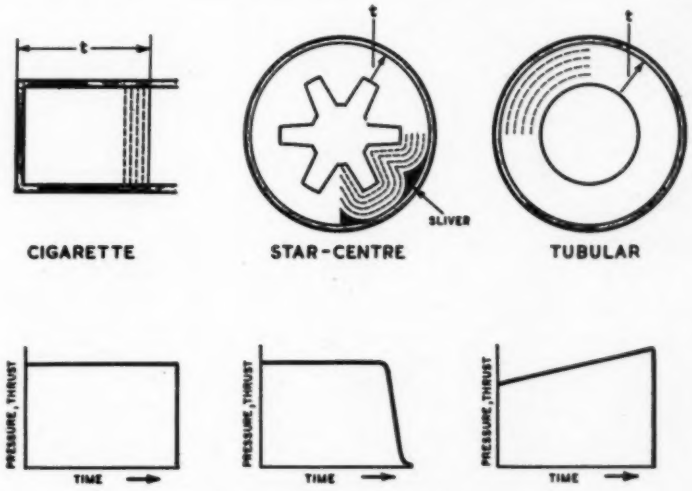


FIG. 7. Simple inhibited charges.

pressure towards the end of burning, which can only be relieved by using a small ratio of propellent thickness to charge diameter, which means that the cross-section of the motor is not filled efficiently, *i.e.*, the loading density is low. The problem is thus to shape the initial burning surface of the charge so that it has the same area as the final surface (which is the same as that of the inner surface of the inhibitor) and maintains this area during burning, without significant variations. This can be done by convoluting the initial inner surface of the charge to a star section,[6] by the use of a number of perforations, or by using a cigarette charge which burns only on the flat surface of one end, all other surfaces being inhibited. The star-centred charge, while being relatively easy to make, does suffer from some disadvantages. It is not, in general, possible to design the initial surface so that the burning surface reaches the surface of the inhibitor at all points simultaneously, so that there is, near the end of burning, a rapid but not complete reduction in the burning surface area. This may reduce the pressure so fast that the remaining propellent is extinguished, which reduces the available total impulse, or there may be a relatively long "tail" formed on the thrust/time curve as the small areas of propellent burn away: such a tail may affect vehicle performance. In addition, it is not easy, in star-centred charges, to make the ratio of the propellent burning thickness to the charge diameter either very large or very small, without affecting either the shape of the thrust/time curve or the loading density. It is for such reasons as these that the more complicated multiperforated charge (Fig. 8) has been developed, which can have small propellent thickness between the perforations without reducing the loading density, and which has little unused propellent at the end of burning.

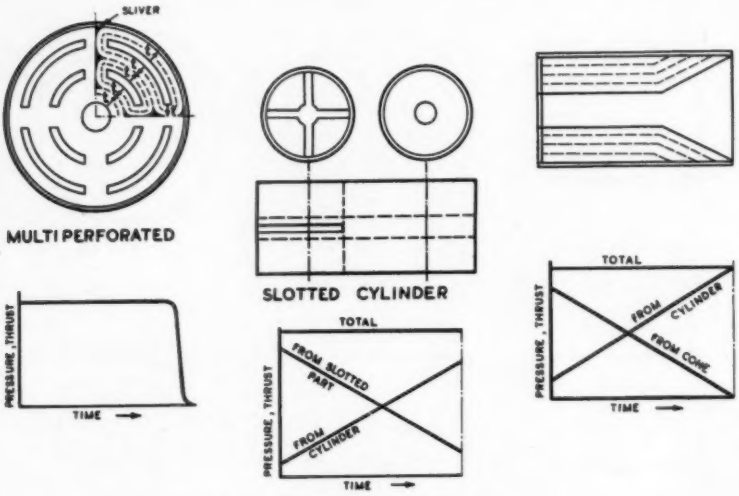


FIG. 8. More complicated inhibited charges.

The externally restricted tubular charge, which is inherently progressive, may be compensated by cutting slots through the propellent nearly to the inner surface of the inhibitor, at one end, or by using a conical surface at one end. Both these methods[6] introduce a rapidly-regressive section which tends to balance the progressive tubular part.

It may be seen that, for different purposes different charge shapes are best suited: for extremely long burning times with low thrusts, the cigarette burner is the obvious choice; for medium thrusts and times, a slot-compensated tubular charge with a small central hole might be used, while for short times and high thrusts a star-centred or multiperforated charge would be chosen. The choice of a propellent with a suitable burning rate at the chosen working pressure enables the charge design to be completed and the nozzle to be designed, using equations (1), (2), and (3), and the usual nozzle equations. The expansion ratio of the nozzle is usually chosen as a compromise between the conflicting requirements of light weight and maximum specific impulse, and it also depends on the height range through which the rocket has to work.

## 5. Performance of Real Charges: Instability

The foregoing discussion has ignored some points which are observed in the performance of real rocket motors, and in particular it has been assumed that a constancy of burning surface area implies constant pressure and thrust. This is, unfortunately, not so: there are two major deviations from the simple state which must be discussed. One is relatively uncomplicated, and is the result of the phenomenon of erosive burning mentioned earlier: if we have, for example

(Fig. 9), an externally inhibited tubular charge whose initial conduit cross-sectional area is only slightly larger than that of the nozzle, the velocity of the gas through the conduit at the rear end of the charge may tend towards the limiting sonic value and so produce considerable erosion, which increases the chamber pressure and hence the burning rate all over the exposed surface of the charge. The erosion diminishes rapidly, since the conduit soon becomes much bigger than the throat, and so we have the appearance of pressure peaks at the start of burning.[7] Such behaviour may be eliminated or reduced by deliberately choosing the initial shape of the conduit (of a star-centred or multiperforated charge) so that the pressure in the absence of erosion would not rise, immediately on ignition, to the general burning level; this permits the erosion to square-off the pressure/time curve. More simply but less efficiently, the erosive peaks can be eliminated by making the conduit cross-sectional area larger and reducing the initial gas velocity.

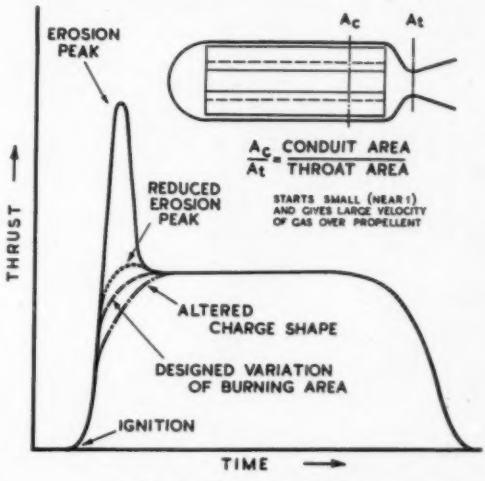


FIG. 9. Erosive burning.

The other deviation is a much more subtle trouble and is also less easy to ameliorate: this is an unsteady burning of the charge, during the normal burning time, which is caused by oscillations of the column of hot gas in the charge conduit. It appears that circular conduits are particularly prone to this trouble,[7,8] and a considerable amount of learned mathematics[8,9] has been done on the modes of these oscillations, but a detailed description of this work would be out of place here. However, hypotheses exist which indicate that there is a coupling between the gas motion, the boundary layer thickness, and the local heat transfer back into the propellent, which may enable oscillations to be produced and to grow to a state of large amplitude and resonance at one or more particular geometrical configurations through which the conduit passes as the propellent burns, producing (by virtue of the non-linear relation between pressure and burning rate) an enhanced total release of gas, which, for a short time, increases the pressure and gives the characteristic peaks of unstable burning (Fig. 10). Cures which have been tried for this instability include the drilling of holes through the walls of uninhibited tubular charges, and the suspending of metal rods down the centres of the conduits of inhibited charges. Both are intended to act as resonance dampers, but the cures are by no means always effective.

## 6. Body Components

The inert components of a solid propellent rocket motor are basically simple, and comprise, generally, a tube, with a large- or full-diameter opening at one end to allow the charge to be introduced, a closure for this opening, and a nozzle through which the combustion gases may expand (Fig. 1). There are also other details, such as the means for applying the thrust to the vehicle, and a charge support if one be needed, and the motor must be provided with an igniter for the charge: this is usually an electrically initiated firework.

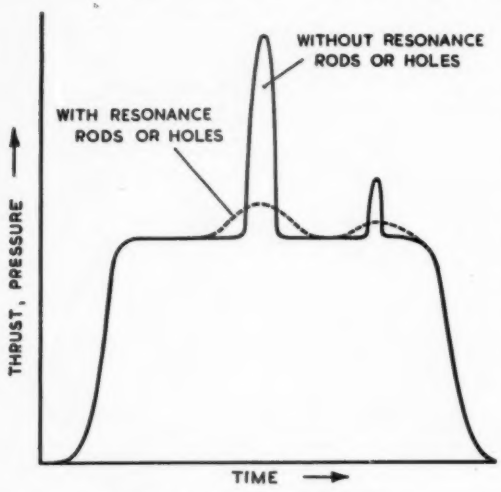


FIG. 10. Resonance in charges.

The body tube, apart from any insulation which may have to be applied to its inner surface, is most frequently made of high-tensile steel, using either welding or manipulative processes, and the nozzle and end closure are usually of steel also.[1] Whereas liquid propellent rocket engines can be cooled by regenerative systems, there are no such possibilities with solid propellent motors unless undesirable complications are introduced, and the design method appears to be one which relies on refractory insulating coatings and thermal sinks. It will be appreciated that at the combustion temperatures which prevail (up to about 2,900° K. for double-base propellents) no normal engineering materials will persist at all, and only very few (beryllium, molybdenum, tungsten and zirconia, for example) of the expensive materials are any good. This means that almost all solid propellent motors suffer from erosion of the nozzle in the throat region, due to the throat material melting and being washed down the nozzle. Adequate metal backing has to be provided to ensure that heat is conducted rapidly away from the surface and that the nozzle retains sufficient strength not to distort under the gas pressure, but the successful design of a nozzle for a motor which has a high gas temperature and a long burning time is indeed difficult.

The general structural design of the tube and fittings is basically that for any pressure vessel, and need not be described here: it is often necessary, however, to work on low margins of strength to give light weight and good overall specific impulse.

## 7. Conclusions

A description has been given of the basic features and methods of design of solid propellent rocket motors. Application of such motors are very numerous, and cannot all be discussed here. It must suffice to mention that solid propellent rockets are in world-wide use, for assisted aircraft take-off, for boosting guided missiles, for upper-atmosphere sounding rockets, and also for the third stage of the Vanguard satellite project.

Their simplicity and reliability, together with the ability to achieve good values of the full-to-empty mass ratio by the elimination of accessories, makes them always worth considering for vehicle propulsion.

## Acknowledgements

The author is indebted to the Ministry of Supply for permission to publish this paper. Crown copyright is reserved in the illustrations.

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# THE CALCULATION OF FUEL DISTRIBUTION IN STEP-ROCKETS*

By Professor M. L. Williams†

SUMMARY

Indicates the changes in Vertregt's analysis[1,2] of the performance of step-rockets that result on considering the dry mass to be composed of three parts: load-carrying structure, non-structural weight, and power plant weight.

In two recent papers,[1,2] Vertregt has investigated the performance of step-rockets, using incidentally an uncommon notation which, however, seems to be somewhat more logical than others in current use. The purpose of this note is to extend his results in such a manner as to bring more explicitly into focus the interaction of the structural and propellent weight.

## Outline of Solution

The problem formulated and solved by Vertregt can be regarded as one of maximizing the payload of a rocket of $N$ steps for a fixed take-off weight and fixed burnout velocity when operating in a gravitationless field with negligible

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air resistance. The "structural" ratios are considered as fixed parameters, as are the velocities of the exhaust gases; the mass ratios are to be adjusted to achieve the maximum payload.

Repeating the more relevant notations and definitions, one has:

$M_n$, the mass of sub-rocket $n$ of a step-rocket,

$m_{bn}$, the mass of the propellents of the $n$th step of a step rocket,

$m_{cn}$, the dry mass of the $n$th step.

$m_p$, the mass of the payload (consistent notation, actually $M_0$)

$r_n$, the mass ratio of sub-rocket $n$.

$$r_n = \frac{M_n}{M_n - m_{bn}} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (1)$$

$R$, the overall mass ratio of a step-rocket of $n$ steps:

$$R = r_1 . r_2 \ldots r_N \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (2)$$

$s_n$, the "structural" ratio:

$$s_n = \frac{m_{cn} + m_{bn}}{m_{cn}} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (3)$$

$p_n$, the payload ratio:

$$p_n = \frac{M_n}{M_{n-1}}, \quad \left( p_1 = \frac{M_1}{M_0} = \frac{M_1}{m_p} \right) \quad .. \quad .. \quad .. \quad .. \quad (4)$$

$$= r_n \frac{s_n - 1}{s_n - r_n}$$

$P$, overall payload ratio:

$$P = p_1 . p_2 . p_3 \ldots p_N = \frac{M_N}{M_0} = \frac{M_N}{m_p} \quad .. \quad .. \quad .. \quad .. \quad (5)$$

Using then the total ideal velocity of a step-rocket of $N$ steps, the sum of the velocity increments, as the restraint condition among the $r_n$,

$$V = \sum^{N} V_n = \sum^{N} c_n \ln r_n \quad .. \quad .. \quad .. \quad .. \quad .. \quad (6)$$

the variational problem, using the usual Lagrange multiplier technique, is thus:

$$\phi(r_n) = P(r_n) + \lambda \Sigma (V_n - c_n \ln r_n) = \text{minimum} \quad .. \quad .. \quad (7)$$

Upon requiring $\partial \phi / \partial r_n = 0$, $n = 1, 2, \ldots N$, there result $N$ equations in the $(N + 1)$ unknowns, $r_n (n = 1, 2, \ldots N)$ and $\lambda$, to which the $(N + 1)$st equation, (6), is added to complete a determinable set for which the solution was shown to be:

$$r_n = s_n - \frac{s_n}{s_1} - \frac{c_1}{c_n} (s_1 - r_1) \quad .. \quad .. \quad .. \quad .. \quad .. \quad (8)$$

with $r_1$ subsequently determined by substitution of (8) in (6).

**An Extension**

In his analysis, Vertregt points out that it is not strictly correct to suppose the mass ratios and "structural" ratios are independent of each other, and it is

upon this point that one can elaborate. Indeed if the $s_n$ are fixed parameters, then from (3):

$$m_{cn} = (s_n - 1)^{-1} m_{bn} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (9)$$

which implies that the dry mass has been taken proportional to the propellent mass.

It seems pertinent to point out that the dry mass is composed of several components, all of which do not necessarily increase in direct proportion to the propellent mass. Specifically one can consider for example that the dry mass contains that of the engine, load and non-load carrying structure and other non-structural components such as guidance equipment, separation equipment and the like. It is for this reason that the term structural ratio has been enclosed in quotation marks, with a better name preferred—even step mass ratio.

For the purpose of the present example, which is to demonstrate the changes in the analysis due to an alternate consideration of the dry mass, rather than to become involved in detailed weight analysis, the dry mass will be considered to be composed of three parts:

1. *Load-carrying Structure*, $m_{cn}^{(1)}$. Qualitatively it may be shown that the weight of the load-carrying structure is mainly affected by the fuel weight. For example, if one supposes that the fuel tanks also form the walls of the rocket, then this part of the structure may be idealized for illustrative purposes as a right circular cylinder of radius $a$ and of wall thickness $h$ and top and bottom plate thickness $h$. The average mass density of the fuel will be $m_{bn}/(\pi a^2 L)$ and the load factor $(1 + k)$, where $k$ is the acceleration in $g$'s. While the critical design condition can be due to either compressive (buckling) stresses in the shell wall from the inertia loads transmitted through the steps, or tension hoop stresses in the cylinder wall due to inertia loading of the fuel, assume that the latter design condition is critical.* The design hoop stress $\sigma_1$ will then be:

$$\sigma_1 = \frac{aq}{h} = \frac{a}{h}(1 + k)\rho_f L = \frac{a}{h}(1 + k)\frac{m_{bn}g}{\pi a^2}$$

and hence the thickness/radius ratio becomes:

$$\frac{h}{a} = \frac{(1 + k)g}{\pi a^2} \cdot \frac{m_{bn}}{\sigma_1} \sim m_{bn}$$

Similarly the stress in the tank bottom, considered as a circular membrane, gives (see, for example, Timoshenko[3]):

$$\sigma_2 = 0.423E\left[\frac{a}{h} \cdot \frac{(1 + k)g}{E} \cdot \frac{m_{bn}}{\pi a^2}\right]^{2/3}$$

and hence this thickness ratio is also proportional to propellent weight,

$$\frac{h}{a} = \frac{1 + k}{E} \cdot \frac{m_{bn}g}{\pi a^2}\left[0.423\frac{E}{\sigma_2}\right]^{3/2} \sim m_{bn}$$

As the tank weight is linearly related to the sum of the thickness ratios, it is seen that one would expect this part of the load-carrying structure to depend

* A more complete analysis, considering also the overall length/diameter ratio, number of stages and complexity of the multiple units is necessary to analyse the actual design criterion.

upon propellent weight. Upon generalizing this proportionality to other load-carrying elements, one would postulate the linear relation

$$m_{cn}^{(1)} \equiv \gamma_{cn}^{(1)} m_{bn} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (10)$$

which is consistent with Vertregt's assumption, (9).

2. *Non-structural weight, $m_{cn}^{(2)}$.* Assuming, for simplicity in this analysis, that the non-structural weight, $m_{cn}^{(2)}$, is proportional to the load-carrying structural weight, $m_{cn}^{(1)}$, and hence by (10) to the propellent weight, there results, also in accordance with (9),

$$m_{cn}^{(2)} \equiv \gamma_{cn}^{(2)} m_{bn} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (11)$$

3. *Power plant weight, $m_{cn}^{(3)}$.* In contrast with the above postulates, however, it is customarily assumed that the weight of an engine is proportional to the thrust produced. Further, the thrust required is proportional to the weight being moved, *i.e.*, $T \sim (1 + k) M_n g$. Under these conditions one is led to postulate

$$m_{cn}^{(3)} \equiv [c_e(1 + k)g] M_n \equiv \gamma_{cn}^{(3)} M_n \quad .. \quad .. \quad .. \quad .. \quad (12)$$

which dependence is different from (9).

This extension of Vertregt's analysis can then be viewed as a generalization of a hypothesis wherein the dry weight is taken proportional to the wet weight alone, to one wherein the dry weight is composed of two primary dependencies, the wet weight and gross weight.

$$m_{cn} = [\gamma_{cn}^{(1)} + \gamma_{cn}^{(2)}] m_{bn} + \gamma_{cn}^{(3)} M_n \quad .. \quad .. \quad .. \quad .. \quad (13)$$

## The Minimum Problem

It is now a simple matter to formulate the variational problem for this new hypothesis.* For simplicity, first substitute (1) and (3) into (4) to obtain the payload ratio in the form:

$$p_n = [1 - (m_{cn}/M_n) - (m_{bn}/M_n)]^{-1} \quad .. \quad .. \quad .. \quad (14)$$

which upon inserting (10)–(12) gives:

$$p_n = [(1 - \gamma_{cn}^{(3)}) - (1 + \gamma_{cn}^{(1)} + \gamma_{cn}^{(2)})(m_{bn}/M_n)]^{-1}$$

Then let:

$$A_n = 1 - \gamma_{cn}^{(3)} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (15)$$

$$B_n = 1 + \gamma_{cn}^{(1)} + \gamma_{cn}^{(2)} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (16)$$

$$\mu_n = m_{bn}/M_n \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (17)$$

So that:

$$p_n = [A_n - B_n \mu_n]^{-1} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (18)$$

Second, note that the mass ratio, (1), can be written as:

$$r_n = (1 - \mu_n)^{-1} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (19)$$

Using (18) and (19) the variational problem now becomes:

$$\phi(\mu_n) = P(\mu_n) + \lambda\{V + \sum^{N} c_n \ln (1 - \mu_n)\} = \text{minimum} \quad .. \quad (20)$$

* M. Goldsmith[4] has treated a somewhat similar problem for a two-step rocket.

with:

$$V + \sum^{N} c_n \ln (1 - \mu_n) = 0 \quad .. \quad .. \quad .. \quad .. \quad .. \quad (21)$$

as the restraint condition. The solution is readily shown to be

$$r_n = \frac{1}{1-\mu_n} = \left[1 - \frac{A_n}{B_n}\right]^{-1} \left\{1 - \frac{c_1}{c_n}\left[1 - \left(1 - \frac{A_1}{B_1}\right)\right]\left[1 - \mu_1\right]^{-1}\right\} \quad (22)$$

or with $(1 - \mu_1)$ or $r_1$, being determined by inserting (22) into (21).

**Special Case of Equal Exhaust Velocities, $c_1 = c_n = c$**

In the special case of constant exhaust velocity, $c$, a formal solution is again possible:

$$\frac{1}{1-\mu_n} = r_n = \frac{1 - (A_1/B_1)}{1 - (A_n/B_n)} r_1 \quad .. \quad .. \quad .. \quad .. \quad (23)$$

then:

$$V = \sum^{N} V_n = c \ln r_1 . r_2 \ldots r_N = c \ln \left\{\left[r_1\left(1 - \frac{A_1}{B_1}\right)\right]^N \prod_{n=1}^{N}\left(1 - \frac{A_n}{B_n}\right)^{-1}\right\} \quad (24)$$

Define the product function as $\bar{S}$.

$$\bar{S} \equiv \prod_{n=1}^{N}[1 - (A_n/B_n)]^{-1} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (25)$$

so that $r_1$ may be found from (24) as:

$$r_1 = [e^{V/c}/\bar{S}]^{1/N} [1 - (A_1/B_1)]^{-1} \quad .. \quad .. \quad .. \quad .. \quad (26)$$

or in terms of the overall mass ratio $R$, $V \equiv c \ln R$, where $R = \bar{S} r_1^N [1 - (A_1/B_1)]^N$

$$r_1 = [1 - (A_1/B_1)]^{-1} (R/\bar{S})^{1/N} \quad .. \quad .. \quad .. \quad .. \quad (27)$$

and

$$r_n = [1 - (A_n/B_n)]^{-1} (R/\bar{S})^{1/N} \quad .. \quad .. \quad .. \quad .. \quad (28)$$

For the case treated by Vertregt, it is easily confirmed that $A_n = 1$ and $B_n = + S_n/(S_n - 1)$ and his results

$$r_1 = s_1 (R/S)^{1/N}$$

$$r_n = (s_n/s_1) r_1 = s_n (R/S)^{1/N}$$

$$S = \Pi s_n$$

are reproduced.

It should be mentioned in conclusion that the constants of proportionality $\gamma_{cn}^{(i)}$ must be chosen carefully, or computed in much greater detail than given here. If the constants are representative, however, the foregoing analysis should be of value in preliminary design.

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# A NOTE ON SPACE TRAVEL IN A GRAVITATIONAL FIELD*

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SUMMARY

The behaviour of clocks during travel in the curved space-time of the general theory of relativity is discussed, and effects observable in journeys in the Solar system or on Earth satellites are examined.

The Dingle–McCrea controvery[1–6,8] has drawn attention to the differences in time as measured by space travellers. So far the discussions have been confined to the flat space-time of the special theory of relativity, but it is more interesting to see what would happen in the curved space-time of the general theory and it is our purpose to investigate this.

Now the fundamental hypothesis of both the special and the general theory is that space-time is metrical. We can understand this concept better if we think of ordinary two- or three-dimensional space. This is also metrical in the sense that we can measure *distances* between neighbouring pairs of points. If we take the familiar rectangular co-ordinate system $x,y$ in a plane the distance $ds$ between the points $(x,y)$ and $(x + dx,\ y + dy)$ is given by $ds^2 = dx^2 + dy^2$. If we take other co-ordinate systems this expression becomes more complicated. For example, with plane polar co-ordinates the distance between the points with co-ordinates $(r,\theta)$ and $(r + dr,\ \theta + d\theta)$ is given by $ds^2 = dr^2 + r^2 d\theta^2$, and this can be reduced to the form $ds^2 = dx^2 + dy^2$ by changing the co-ordinate system taking $x = r\cos\theta$, $y = r\sin\theta$. With an arbitrary co-ordinate system we shall get a general quadratic form $ds^2 = ad\xi^2 + bd\eta^2 + 2hd\xi d\eta$ for the two-dimensional metric. One way of investigating the geometry would be to start with an arbitrary co-ordinate system, to find the metrical form, and to try to find a change of the co-ordinate system which would reduce the metric to the simplest form $ds^2 = dx^2 + dy^2$. When one has done this one has a rectangular co-ordinate system. This process is virtually the process of triangulation which a surveyor is obliged to resort to when he cannot measure the whole area but only small pieces. Now on the surface of a sphere using spherical polar co-ordinates the metrical expression is $ds^2 = a^2(d\theta^2 + \sin^2\theta d\phi^2)$. There is no change of the co-ordinates which will reduce this to the form $ds^2 = dx^2 + dy^2$. This gives us the clue as to the difference between flat and curved spaces. For flat spaces, planes, there does exist a co-ordinate system which gives $ds^2 = dx^2 + dy^2$, but on curved surfaces the metrical form may be simplified by special co-ordinate systems but will never reduce to just $ds^2 = dx^2 + dy^2$.

Now in our four-dimensional space-time the metrical distance $ds$ will be given by a general quadratic form $ds^2 = \Sigma_{i,k}\, g_{i,k}\, dx_i\, dx_k$, the summation being taken over all four co-ordinates $x_1$, $x_2$, $x_3$ and $t = x_4$. In the special theory

* Manuscript received 24 April, 1957.

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we also assume that a co-ordinate system exists in which this reduces to $ds^2 = -(dx^2 + dy^2 + dz^2) + c^2dt^2$ or to $ds^2 = -(dx^2 + dy^2 + dz^2) + dt^2$ if we measure time in relativistic units, taking $t$ for $ct$, and this special space-time is described as flat in analogy with the case $ds^2 = dx^2 + dy^2$ discussed above. It is a basic hypothesis* of both the special and general theory that a moving clock will measure this metrical distance along its path in a way similar to the way a map-measuring wheel marks off distances along the path it follows, but with differences arising from the negative signs in the metric.† Because the time-like intervals along a specified path can be measured in this way by clocks we often refer to such distances as *proper times*. They are in general quite distinct from the corresponding *co-ordinate times* associated with any arbitrary system of space-time co-ordinates.

Now if we consider two different paths $A\,X\,B$ and $A\,Y\,B$ in space-time the proper times will resemble the metrical distances of two different paths $A'X'B'$ and $A'Y'B'$ in ordinary space. Now we all know that the path length depends on the path chosen. It is in general different for different paths and the straight path is the shortest path. So in space-time will the proper times of two paths seldom be the same and the proper time of the straight path or geodesic would be, with positive metric, the least. With negative terms in the metric the interval $ds^2$ is decreased, not increased, by introducing $dx^2$, $dy^2$ or $dz^2$ and the proper time of the geodesic is greater, not less, than that of other paths. This discussion shows that Dingle's conclusions are in obvious contradiction with the hypothesis as to what clocks measure and since he apparently accepts the hypothesis his argument must be false. This is, however, a diversion from our purpose, which is to discuss proper times in a curved space-time.

Now the analogy between curved space-time and a curved two-dimensional space is valid if we allow for the effects of negative signs in the metrical expression. Let us therefore investigate the properties of a curved space-time by comparing proper times with distances on the surface of a sphere. The shortest paths or geodesics are then parts of great circles. Now the great circle path from, say, London to Cardiff is shorter than the London, Edinburgh, Cardiff path, but there is another great circle path from London via New Zealand to Cardiff and this is longer than the London, Edinburgh, Cardiff path. It follows that in a curved space-time the geodesic paths are not necessarily the shortest for positive metrics or the longest for metrics with negative terms. It is, of course, possible that some of the alternative geodesic paths may be eliminated by the condition that clocks must not travel faster than light and all parts of the path must be time-like, but this leads to questions of topology which are beyond the scope of this article. All we can say with certainty is that the geodesic path lengths are stationary for small deformations of the paths. Our space traveller might come back older instead of younger than his earthbound twin.

* See Note 1 of Appendix.

† See Note 2 of Appendix.

**Application to the Solar System**

The most interesting problem is to compare different paths within the solar system and to see that travellers to Neptune may come back older, while travellers to Venus will come back younger than their earthbound twins. This we can do by assuming that the space-time round the Sun is described by the Schwarzschild metric

$$ds^2 = -(\gamma^{-1}dr^2 + r^2d\theta^2 + r^2\sin^2\theta\, d\phi^2) + \gamma dt^2, \quad \gamma = (1 - 2m/r)$$

Here $r$, $\theta$, $\phi$ and $t$ are a special co-ordinate system. Since the metric is independent of $t$, light signals travelling between two points fixed in the space co-ordinates will always take the same time and the time separation of two such signals will be the same at reception as at emission. We can therefore compare changes of co-ordinate time at different points and $t$ will behave in some ways like an absolute time. Clocks stationary in the co-ordinate system will, however, not read this time but their own proper times $ds$ given by $ds^2 = \gamma dt^2 = (1 - 2m/r)dt^2$. Thus we may say that compared with the co-ordinate time they go at different rates $ds/dt = \gamma^{\frac{1}{2}} = (1 - 2m/r)^{\frac{1}{2}}$ which depend on their distance from the Sun. This formula enables us to compare them indirectly with one another without actually determining $t$.

Clocks which are moving freely instead of being fixed in the co-ordinate system have different values for $ds/dt$. The equations of the geodesics have been calculated in Eddington's "The Mathematical Theory of Relativity,"[7] § 39. This also uses $t$ for $ct$ and with this relativistic scale for time Eddington's equation 39.42 gives:

$$\frac{ds}{dt} = \frac{\gamma}{C} = \frac{(1 - 2m/r)}{C} \qquad \ldots \quad \ldots \quad \ldots \qquad (1)$$

where $C$ is a constant to be obtained from the metrical expression. This, taking paths in the plane $\sin\theta = 1$ gives

$$\left.\begin{aligned} \left(\frac{\gamma}{C}\right)^2 = \left(\frac{ds}{dt}\right)^2 = \frac{1}{dt^2}\left[-\gamma^{-1}(dr)^2 - r^2(d\phi)^2 + \gamma(dt)^2\right] = \gamma - \gamma^{-1}\dot{r}^2 - r^2\dot{\phi}^2 \\ \frac{1}{C} = \gamma^{-1}\,(\gamma - \gamma^{-1}\dot{r}^2 - r^2\,\dot{\phi}^2)^{\frac{1}{2}} \end{aligned}\right\} \qquad (2)$$

Now for a comparison with classical mechanics we write

$$T = \tfrac{1}{2}(\dot{r}^2 + r^2\dot{\phi}^2), \quad V = -m/r$$

$T$ is then the kinetic energy per unit mass of the clock or space-vehicle carrying the clock measured in relativistic units. In ordinary units this is the kinetic energy divided by the rest mass energy, i.e., mass $\times$ (velocity of light).[2] When $T$ and $V$ are small we can ignore second order quantities like $TV$ and obtain

$$\frac{1}{C} = (1 + 2V)^{-1}\,(1 + 2V - 2T)^{\frac{1}{2}} \cong 1 - V - T = 1 - V_0 - T_0$$

This is just the energy conservation equation and we see that $V$ is just the potential energy per unit mass measured in the same units as $T$ or the potential energy measured in ordinary units and divided by the rest mass energy. The equation for the proper times then becomes

$$\frac{ds}{dt} = \frac{\gamma}{C} = (1 + 2V)/C \cong 1 + V - T = 1 + 2V - V_0 - T_0$$

Now the interpretation of this last result is very important. Since there is no absolute time against which to compare the clock proper time, the statement that a clock slows down has no corresponding absolute meaning. Because of this the physicist finds it introduces no ambiguity to use the phrase "slowing down" to describe the comparison of a clock proper time with a particular co-ordinate time. Unfortunately, this may lead the philosopher to think he is talking about an absolute change in the clock, which is not the case. He is talking of the comparison of a proper time with the projections $dx$, $dy$, $dz$ or $dt$ upon a co-ordinate system. Some may say that the projection changes $ds$, but they only mean that it creates a new entity $dt$ which is different from $ds$ and the ratio $ds/dt$ is only a projection ratio like $\sec\theta$ in ordinary space. Thus $ds/dt \neq 1$ does not imply a change in the clock measuring $s$ but only that we are comparing different entities. Nevertheless, using the phrase "slowing down" to describe $ds/dt < 1$ has a useful meaning when we wish to compare the proper times of two clocks which have travelled different paths with the same starting and finishing points in space-time. For then $\Delta s_1 = \int_A^B (ds_1/dt)dt$ and $\Delta s_2 = \int_A^B (ds_2/dt)dt$ and if $ds_1/dt$ is consistently less than $ds_2/dt$ we shall have $\Delta s_1 < \Delta s_2$. The first clock's estimate of the proper time of its journey will be less than the second clock's estimate for a different journey. The result is as though the first clock had been, in ordinary language, running slower than the second, though the basis of comparison is a particular co-ordinate time and not an absolute time. Now for paths which do not end together this comparison is not so useful, because then $\Delta s_1 = \int_A^B (ds_1/dt)dt$, $\Delta s_2 = \int_A^C (ds_2/dt)dt$ and the fact that $B \neq C$ makes the comparison more complicated. Though $\Delta s_1$ and $\Delta s_2$ are not changed by change of the co-ordinate system the time projections $AB$ and $AC$ are, and the ratio $ds_1/dt$ to $ds_2/dt$ is therefore dependent on the co-ordinate system and hence cannot measure an absolute ratio of the rates of goings of the clocks $s_1$, $s_2$. We can see this in another way if we realise that $ds_1$ and $ds_2$ are the sections of two paths lying between the particular co-ordinate values $t$, $t + dt$. If we change the orientation of our co-ordinate axes this sectioning changes the ratio of $ds_1$ and $ds_2$. It follows that the comparison of $ds_1/dt$ and $ds_2/dt$ is only useful in the case where the paths start and finish together. This is the case which Dingle and McCrea discuss and we have shown that there is a meaning in saying that one clock is running slower than another, this meaning being that $ds_1/dt < ds_2/dt$ and implying $\Delta s_1 < \Delta s_2$ for coterminal paths.

We may now say that our clocks are slowed down if they have positive kinetic energy $T$ or negative potential energy $V$. The proportionate slowing down is equal to the total kinetic and negative potential energy $T + |V|$ per unit mass measured in relativistic units or to the total positive kinetic and negative potential energy per unit of rest mass energy.

For the case $m = 0$, $\gamma = 1$, equations (1) and (2) give

$$ds/dt = 1/C = (1 - \dot{r}^2 - r^2\dot{\phi}^2)^{\frac{1}{2}} = (1 - v^2)^{\frac{1}{2}}$$

and this gives us a comparison between proper time and co-ordinate time for clocks moving in a flat space.* It has the same interpretation that moving clocks run slower with increasing kinetic energy but gives a more exact expression for the effect when the relative speeds are not small. The speeds are, of course, speeds relative to a co-ordinate system in which the metric has the special form $ds^2 = -(dx^2 + dy^2 + dz^2) + dt^2$, a condition which is usually described by saying they are speeds relative to an inertial co-ordinate system as our metrical condition makes the space co-ordinate origin describe a geodesic. A clock stationary in such a system also describes a geodesic and has a greater value for its proper time than any other clock which starts and finishes at the same points in space-time.

**Some Numerical Results**

Now Eddington calculates that the gravitational part $V$ or $m/r$ of the slowing down amounts to 2 parts in $10^6$ at the surface of the Sun, and consequently on Earth (at 200 times as far from the Sun's centre) it will amount to 1 part in $10^8$. The kinetic energy for circular orbits is in magnitude half of the (negative) gravitational energy, and consequently the total slowing of clocks on Earth amounts to $1\frac{1}{2}$ parts in $10^8$. On Venus it would be $2\frac{1}{4}$ parts, on Mars 1 part and on Neptune 1/20 part, all in $10^8$. We deduce therefore that space-travellers with an orbit with aphelion near Neptune's orbit will spend a large proportion of their journey at large distances where $|V|$ and $T$ are small and their clocks will run faster than Earth clocks and they will come back a little older, but the maximum ageing in any outward gravitational orbit as compared with Earth clocks will not amount to more than $1\frac{1}{2}$ parts in $10^8$. On the other hand, the maximum rejuvenation of voyagers near the Sun will not exceed 2 parts in $10^6$ from gravitational effects and is not likely to exceed a similar amount from kinetic effects.

Gravitational effects in the solar system cannot change $T$ by more than the change in $|V|$ and this, as we have seen, is trivial. Other means would have to be adopted to achieve velocities comparable with that of light or kinetic energies comparable with the rest mass energy. Even nuclear fission motors would find this difficult as they produce only 200 MeV. from the fission of a uranium atom whose rest mass energy is a thousand times greater. The motor would have to concentrate the energy from fuel a thousand times the weight of the rest of the rocket and yet have enough matter to shield the passengers from gamma radiation with heavy shields and preserve them from the great heat liberated.

As to the gravitational effects of the Earth itself, compared with the Sun its mass is reduced by $3 \times 10^{-6}$ and its radius by $10^{-2}$, and hence the effect of Earth's gravitation may be expected to be about 7 parts in $10^{10}$. For satellites in a close Earth orbit their kinetic energy should give an effect half of this and they will suffer the same gravitational slowing as Earth clocks. Consequently a measurement would not check the gravitational slowing which we would like to do to find whether the general theory applies to our universe.

* See Note 3 of Appendix.

If, however, we can send a clock to the Moon or to a similar orbit and observe its time by radio or light signals we should find it subject to only 1/60 of the Earths gravitational effect (plus half as much again from kinetic energy) and so since this is negligible the Earth clocks should go slower than Moon clocks by 7 parts in $10^{10}$. If this could be measured it would confirm the general theory of relativity.

## APPENDIX

NOTE 1.—The conflict between the results of the Michaelson–Morley experiment and astronomical aberration finally forced the abandonment of the idea that there was an absolute time which all clocks measured. The metrical hypothesis seems to give the only other thing they could measure. Also the statement of what clocks measure is implied by and gives physical meaning to our statement that space-time is metrical, which would be meaningless if we could not measure metrical distances. We may if we like limit the hypothesis to apply only to uniformly or freely moving clocks and to apply the idea to a general path we should have to approximate to it by a succession of clocks freely moving with appropriate velocities. This is what Dingle has done in one paper[2] though he needs only two clocks for that particular case. The process is analogous to measuring the length of a curve by rectifying it and adding the lengths of the small straight parts. The point is, however, irrelevant to most of the above discussion, which is concerned with free clocks describing geodesics.

NOTE 2.—Clocks can only travel with a velocity less than that of light and this gives $dx^2 + dy^2 + dz^2 < dt^2$ or $ds^2 > 0$. The other sort of intervals, for which $dx^2 + dy^2 + dz^2 > dt^2$, are measured by moving measuring rods, but this is irrelevant to our argument.

NOTE 3.—Here we see one of the flaws in Dingle's argument. $ds/dt = (1 - v^2)^{1/2}$ gives a slowing down of the moving clock on both its outward and return journeys, in contradiction to Dingle's statement.

NOTE 4.—The name *Relativity* may seem odd to some readers when they find that the basic hypothesis is that the concepts of absolute space and absolute time are to be replaced by the concept of a unified metrical space-time. The name, however, must be taken as referring to the consequent relativity of time (and also of space). It is one of the main points of our discussion that clock readings or proper times are dependent on the motions of the clocks and are therefore not measurements of something absolute.

Newtonian mechanics gave a relativity of motion in the sense that the equations $m\ddot{x} = X$, etc. are unchanged when we go to a uniformly moving observer. There is no preference for any of the possible inertial co-ordinate systems moving uniformly with respect to one another. It was essential that this property be preserved in going from Newtonian mechanics to a new mechanics and the insistence on its preservation by some early writers might confuse some philosophers into thinking that the relativity of motion was a new feature of the new theory of Relativity and a reason for the name but this cannot be so when Newtonian mechanics satisfy the condition also.

The special theory as stated above has many justifications in experiment which do not need to be stated here. It is the extra change of the metrical form which will describe an effect similar to gravitation (cf. $T + V =$ constant with $V = m/r$ in special units), but with very small extra changes such as our clock effect that still requires experimental justification to decide whether we live in a flat space-time with gravitation as an extra feature or whether the planetary orbits are just features of a curved space-time.

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# THE EFFECTS OF ACCELERATION ON THE HUMAN FRAME

*(Report* of a lecture by Wing Commander F. Latham given to the Society on 2 February, 1957.)*

Wing Commander Latham, of the Royal Air Force Institute of Aviation Medicine at Farnborough, began his lecture by saying that, rather than attempt to cover the whole field and include matter which was already familiar to students of the subject, he would treat it like a buffet supper, taking morsels here and there.

It is no use just asking how much acceleration the human frame can tolerate, without specifying how the acceleration is applied; there are six factors which affect human tolerance: magnitude, duration, direction, site, area, and rate of application of the accelerating force. A convenient mnemonic for this list is: "Mother Drinks Daddy's Sherry And Rejoices."

If the duration of acceleration is short, the effect is merely mechanical, as if applied to a sack of potatoes. What is most important is the *rate of change* of acceleration. As to sudden stoppages of motion in a straight line, i.e.,

[*Air Ministry photograph*

FIG. 1. Rocket propelled trolley with dummy *in situ*, at the position of impact with the decelerating ram.

* Report prepared by Dr. A. E. Slater and revised by the lecturer.

linear decelerations, the effect on the body depends on its supporting harness. Under crash conditions, restraint is not applied to the head, and the result is that the head is jerked forward and damage may be sustained by the muscles at the back of the neck, or even by the larynx in front if much clothing or equipment is worn in this region.

In recent years it has been shown that the human body can tolerate very high deceleration loads if properly supported. For research and experimental purposes, a deceleration of 12 *g* with a rate of change of 180 *g* per sec. has been found quite adequate for the physiological testing of harness systems.

A film was shown by W. Cdr. Latham which demonstrated in slow motion what happened to a body when abruptly decelerated on a rocket-driven trolley. The deceleration was effected by a cylinder of water on the trolley impinging on a stationary hydraulic ram fixed at the end of the track; the covering of the front end of the water cylinder was broken upon contact with the ram. A typical deceleration was 12 *g*, lasting for 0·3 or 0·4 sec.

Upward acceleration in an ejection seat is another problem investigated at the Institute, with the result, the lecturer claimed, that the British type of ejection seat had proved itself better than that of any other country. The seat has to achieve a velocity of 60 ft./sec. over a distance of about 42 in. only; this requires some 20 *g* to be applied at a rate of 250 *g* per sec. The man plus seat behaves as a double mass spring coupled system, with the result that when the acceleration is first applied, there is a lag in the movement of the man. This leads to an acceleration overshoot acting on the man's body. In fact the "spring" in the system has a natural period of oscillation which can only be varied by changing the properties of the cushion, as the human body cannot be changed. The dynamic characteristics of the ejected system

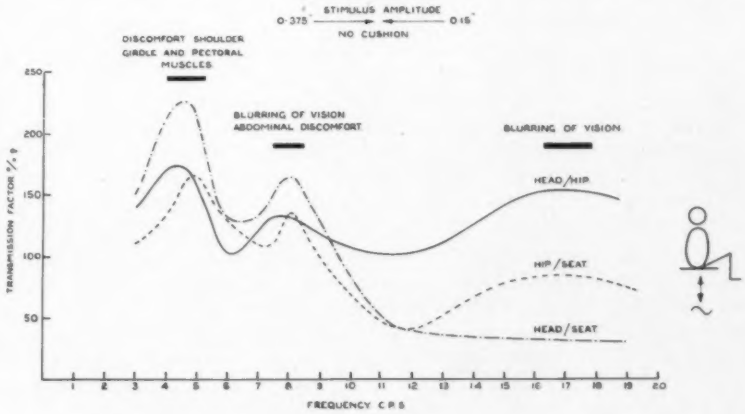


FIG. 2. Frequency resonance curve for the seated body. The ratios of the accelerations experienced by the seat and body during sinusoidal vibration are termed the transmission factor.

having been defined in terms of natural frequencies and damping factors, it has been possible to investigate the problem with the aid of an analogue computor.

The body, in fact, has several natural periods of oscillation, the most important being those of 5, 8 and 17 cycles per sec. The last is the resonant frequency of the head on the neck. At approximately 70 cycles per sec. the eyeballs oscillate in their sockets and interfere with vision. These findings, W. Cdr. Latham said, are of use not only in aviation, for car designers have shown interest in them.

In the case of long duration accelerations, physiological experiments are carried out on the human centrifuge. This machine, which applies acceleration in the form of "centrifugal force," measured in units of *g*, is well known to students of spaceflight problems. That at the Institute of Aviation Medicine was described by W. Cdr. Latham; it is of 30 ft. radius, powered by a 2,200 horse-power D.C. motor, and can apply 30 *g* in 9 sec. A film illustrated the operation of the centrifuge.

At the discussion following the lecture, the first question related to the findings from experiments with this centrifuge. What is the maximum *g* tolerance if the acceleration is continued indefinitely? Only about 2 *g* or 2¼ *g*, the lecturer replied. But 3 *g* can be withstood for 15 min., 5 *g* for 5 sec., and possibly 15 *g* for 1½ sec. In the prone position 12 *g* can be tolerated for a minute or so, but the eyelids become too heavy for their muscles to lift them and respiration is difficult. Would an "iron lung" help respiration in the prone position, when it becomes difficult owing to excessive *g*? Not significantly, the lecturer thought; the body would be nearing its physiological limit anyway.

In spite of the various limits found in physiological experiments on the centrifuge, very high *g* forces can be tolerated if they only last for a very brief period. The *g* loads on the body during outdoor games can be surprisingly high. When walking along a concrete floor, for instance, one's ankle joint is being subjected to peak loads of at least 7 *g* at each step. Another striking example given by W. Cdr. Latham was that of a woman who attempted suicide by jumping from a fourth-floor window; she was unsuccessful because she fell flat on to freshly turned earth. It was calculated, from the depth of the hole she made, that she must have suffered a deceleration of 150 *g* during a small fraction of a second.

Can drugs increase anyone's tolerance to excessive *g*? They might produce a slight improvement, the lecturer thought, but so slight that it was not worth while using them. How about anti-*g* inflatable trousers? They give added protection of up to the equivalent of 2 *g*. Could a man be conditioned to a higher *g* level than normal? This question has been investigated, but the answer is "No"; if anything, repeated exposure reduces tolerance, though less so in the young than in older people.

At the conclusion of the discussion W. Cdr. Latham was thanked heartily for his most interesting lecture.—A. E. S.

# ROCKET ENGINE DATA SHEETS*

COMPILED BY D. S. CARTON, A.F.R.Ae.S.

The compiler wishes to thank the Ministry of Supply for permission to publish details of the R.A.E./R.P.D. Beta 1 rocket engine.

**R.A.E./R.P.D. Beta 1 Rocket Engine.** Main engine for vertical take-off model. Fairey V.T.O. model.

SEA LEVEL MAXIMUM PERFORMANCE

Thrust: 1,800 Lb. ± 5 per cent.
Propellant flow rate: 10·15 lb./sec.
Specific propellant consumption: 21·5 lb./hr./Lb.
Specific impulse: 177 Lb.sec./lb.
Effective exhaust velocity: 5,700 ft./sec.
Time of operation: 48 sec.

(Performance figures above include 0·65 lb./sec. hydrogen peroxide.)

*Propellants.*—Oxidant: Hydrogen peroxide 80 per cent. with water 20 per cent. (7·63 lb./sec.). Fuel: Methanol 57 per cent., hydrazine hydrate 30 per cent., water 13 per cent. plus trace of copper base catalyst (2·52 lb./sec.). Effective oxidant/fuel ratio in chamber 2·78 : 1.

DESCRIPTION

*Tanks.*—Oxidant and fuel contained in polyvinylchloride bags and expelled to pumps by air or nitrogen pressure at 3 Lb./in.$^2$ during operation.

*Feed.*—Two pumps driven by turbine at 24,000 r.p.m. Pumps, centrifugal Barske-type: 2·4 in. diameter with carbon bearings. Impulse turbine: 3·8 in. diameter. Steam generator, peroxide forced through cylinder containing 145 silver-plated gauze discs about 1·8 in. diameter.

*Combustion chambers.*—Two internally identical chambers. Operating pressure: 240 Lb./in.$^2$ Diameter: 6 in. L* 100 in. Nozzle throat: 1·75 in. diameter with 15° expansion half-angle. Overall length: 12·7 in. Chamber specific impulse: 190 Lb.sec./lb.

*Injectors.*—Fuel, ring of 20 swirl orifices on flat end face. Peroxide orifices in side wall, flow across end face produced by *target plate*. One steam injector assists ignition.

*Cooling.*—Regenerative circulation of peroxide, round chamber and nozzle from throat only. Helical flow produced by spiral spacer strips.

*Ignition.*—No ignition system as peroxide reacts spontaneously with hydrazine hydrate in fuel.

*Starting.*—Initial rotation of turbine produced by an external supply of peroxide to steam generator *via* a slip coupling. Main propellant lines are thus pressurized and also bled. Observer then switches to main flow and unit fires.

*Weight.*—Not known.

* *Units.* The abbreviation Lb. is used for pound weight.

*Operational details.*—Installed in the Fairey Vertical Take-off model. Directional control at low forward speeds obtained by thrust line deflection Combustion chambers are permitted to swivel, one at right-angles to the other.

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**Reaction Motors Inc.: 6000 C4 Rocket Engine.** Main engine for aircraft. Used in the Bell XS-1.

SEA LEVEL MAXIMUM PERFORMANCE

Thrust: 6,000 Lb.
Propellant flow rate: 30 lb./sec.
Specific propellant consumption: 18 lb./hr./Lb.
Specific impulse: 200 Lb.sec./lb.
Effective exhaust velocity: 6,440 ft./sec.
Time of operation: 2·7 min. in flight.

*Propellants.*—Oxidant: liquid oxygen. Fuel: ethanol 75 per cent. with water. Effective oxidant/fuel ratio: 1·3 : 1.

DESCRIPTION

*Tanks.*—Not part of unit. Material: Stainless chrome molybdenum steel.

*Feed.*—Nitrogen pressurization. Stored as a liquid in a 36-in. diameter sphere of 3-in. thickness. Stainless-steel bottle and evaporator tested to 9,000 Lb./in.$^2$

*Combustion chambers.*—Operating pressure 230 Lb./in.$^2$. Four separate, identical chambers grouped around main beam support. Welded construction.

*Cooling.*—Regenerative circulation of fuel, maybe small use of evaporative cooling off walls. Outer wall temperature 60° C.

*Ignition.*—Separate, spark-plug ignited bi-propellant igniter chambers screwed into each main chamber. Main propellant flow withheld until satisfactory igniter pressure obtained.

*Control.*—Cockpit control permits selection four, or less chambers in steps of 1,500 Lb. thrust. No provision is made for continuously variable thrust.

*Weight.*—Quoted as 210 Lb. This appears to exclude the pressurization system, evaporator and tanks.

*Operational details:* First powered flight made in Bell XS-1 in December, 1946, at Muroc Field. Original project for the X-1 called for turbo pumps, but these were not available in time. The temporary pressurization system put up landing weight by 2,000 Lb., decreased the propellant load carried, and so decreased firing time by 1·5 min. Alternative engine designation: XLR–11–RMI. More recent installations include hydrogen peroxide reactants driven turbo-pumps. Douglas Skyrocket, and Bell X–1A and X–1B.

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**B.M.W. 109-510 Main Aircraft Rocket Engine.** Controllable thrust engine. Intended for Me 163 B.

SEA LEVEL MAXIMUM PERFORMANCE

Thrust: 3,300 Lb.
Propellant flow rate: 18·3 lb./sec.
Specific propellant consumption: 20·6 lb./hr./Lb.
Specific impulse 175 Lb.sec./lb.
Effective exhaust velocity: 5,640 ft./sec.
Time of operation 8–15 min. normal.

*Propellants.*—Oxidant: Nitric acid 98–99 per cent. concentration. Fuel: Methanol. Effective oxidant/fuel ratio 2 : 1.

DESCRIPTION

*Feed.*—Two pumps driven by turbine. Impellors, centrifugal vane type. Gas producer, controlled reaction of nitric acid and a methanol water mixture. Pump delivery pressure 600 Lb./in.$^2$

*Combustion chamber.*—Operating pressure 515 Lb./in.$^2$ Maximum diameter 5·6 in. L* 75 in. Exhaust nozzle 2·25 in. diameter with 12° expansion half-angle. Total weight 55 Lb. Inner shell, chrome nickel steel.

*Injectors.*—Impinging streams. Eight holes in hollow head.

*Cooling.*—Regenerative circulation of nitric acid through two coolant passages between inner and outer shells.

*Ignition.*—Small electric pump injects a liquid which is self-igniting with nitric acid.

*Control.*—Thrust controlled hydraulically. Maximum thrust limited by combustion chamber pressure sensing control.

*Weight.*—440 Lb.

*Operational details.*—The original order for this unit was given in 1943 due to the troubles being experienced with the operation of the Walter 109–509 A1 in the Me 163 B. In its turn the development of the B.M.W. unit was held up due to difficulty with the nitric acid/methanol reaction in the gas producer for the turbine. Tests were finally made using the Walter *steam pot* and turbine to drive B.M.W. pumps. The combustion chamber reached a very satisfactory development and was then used for the B.M.W. 109–718. Alternative designations for the 510 are B.M.W. 3390 A, and R. II 303.

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## WALTER CONTROLLABLE THRUST ROCKET ENGINE SERIES

**R II 203 Rocket Engine.** Main engine for Lippisch D.F.S. 194 and Me 163 prototypes.

SEA LEVEL PERFORMANCE

Thrust: 1,650 Lb. Specific impulse: 106 Lb.sec./lb.

*Propellants.*—Monopropellant system. Hydrogen peroxide 80 per cent. with 20 per cent. water. Catalyst liquid, calcium permanganate.

DESCRIPTION

*Feed.*—Two pumps driven by turbine. Steam for turbine generated by decomposition of peroxide sprayed on to catalyst stones (foamed cement treated with permanganate) in Walter *steam pot.*

*Combustion chamber.*—Operating pressure 265 Lb./in.$^2$ at maximum thrust. Minimum thrust 330 Lb. Chamber not cooled.

*Weight.*—167 Lb. complete, less tanks.

*Operational details.*—Design started in 1939. First flown in 1941 in D.F.S. 194. Used as a basic research engine, and for pilot handling experience. From the R II 203 was developed the R II 211 bipropellant engine series.

**109-509 AO Rocket Engine (R II 211 series).** Main engine for Me 163 B.

SEA LEVEL MAXIMUM PERFORMANCE (average figures—*see* end note)

Thrust: 3,310 Lb. Specific impulse: 183 Lb.sec./lb.
Propellant flow rate: 18·1 lb./sec. Effective exhaust velocity: 5,900 ft./sec.
Specific propellant consumption: 19·6 lb./hr./Lb. Time: 4 min. (tank limit).

(Performance figures above include 0·9 lb./sec. hydrogen peroxide flow to drive turbine.)

*Propellants.*—Oxidant: Hydrogen peroxide 80 per cent. with 20 per cent. water. Fuel: Methanol 57 per cent., hydrazine hydrate 30 per cent., water 13 per cent., plus trace of potassium cuprocyanide catalyst. Average combustion oxidant/fuel ratio 3 : 1.

DESCRIPTION

*Feed.*—Two pumps driven by turbine. Steam from Walter *steam pot* as in R II 203 above. Peroxide flow rate 0·9 lb./sec. $\pm$ 5 per cent.

*Turbine and pump assembly.*—Single shaft 16,600 r.p.m., design maximum. 68 Lb. total. Impulse turbine: 7·8 in. mean diameter, high-tensile steel disc with single row of separate blades. Develops 90 h.p. Pumps: both fully shrouded centrifugals of aluminium with screw-type priming stage. Fuel: 3·9 in. diameter, flow 5 lb./sec. at 540 Lb./in.$^2$ Oxidant: 3·3 in. diameter, flow 16·5 lb./sec. at 570 Lb./in.$^2$

*Combustion chamber.*—Operating pressure 294 Lb./in.$^2$ with combustion specific impulse 193 Lb.sec./lb. at maximum. Minimum 330 Lb. thrust at

51 Lb./in.² pressure with overall specific impulse 75 Lb.sec./lb. Chamber almost spherical, maximum internal diameter 10 in. L* 66 in. Nozzle throat 3·35 in. diameter with 15° expansion half angle. Inner section, mild-steel forging.

*Injectors.*—Twelve swirl injectors with spring-loaded centre pintles, screwed into and equi-spaced upon flat injector head. In each injector fuel forms outer cylinder with inner cone of peroxide spraying upon it. Injector head bolted on to chamber end.

*Cooling.*—Regenerative circulation of fuel. Maximum outlet temperature 100° C.

*Ignition.*—No ignition system used, as hydrazine hydrate spontaneously reacts with peroxide.

*Starting .*—Electric motor, coupled through *free-wheel* arrangement, drives pump assembly. About 90 Lb./in.² pressure is generated in the peroxide, enough to make Walter *steam pot* operate and start driving turbine.

*Control.*—Single pilot's lever gives continuous variation of thrust between 3,310 Lb. and 330 Lb. Compound valve arrangement decreases propellant flow to groups of three injectors, maintaining pressure drop across remainder until only six in operation.

*Weight.*—368 Lb. for complete power plant.

*Operational details.*—Unit developed from R II 203. Pumps, turbine and valves assembled around a cast frame on to which is fixed a thrust tube with the combustion chamber at one end.

**109-509 A1 Rocket Engine (R II 211 series).** Main engine for Me 163 B. (As AO except as follows.)

SEA LEVEL MAXIMUM PERFORMANCE (average figures—*see* end note).

Thrust: 3,750 Lb. Specific impulse: 186 Lb.sec./lb.
Propellant Flow Rate: 20·1 lb./sec. Effective exhaust velocity: 5,990 ft./sec.
Specific propellant consumption: 19·3 lb./hr./Lb. Time: 4 min. (tank limit).
(Performance figures above include 1·0 lb./sec. hydrogen peroxide flow to drive turbine).

DESCRIPTION

*Combustion chamber.*—Operating pressure 324 Lb./in.² with combustion specific impulse of 196 Lb.sec./lb. at maximum. Minimum 474 Lb. thrust at 66 Lb./in.² with overall specific impulse of 87 Lb./lb./sec.

*Injectors.*—Injector head welded into chamber.

*Weight.*—370 Lb. for complete power plant.

*Operational details.*—Pumps and valves assembled within a fabricated frame. The unit was developed from the A0 and was first flown in 1942. Considerable development troubles experienced including hard starting, also pump cavitation under low or negative *g* produced temporary starvation

of one or both propellants followed by high flow rates and explosive mixing. *Fatigued* aluminium joints were found to cause leaks and fires. These problems were not solved quickly and in 1943 B.M.W. were asked to produce an engine for the Me 163 B (*see* data sheet B.M.W. 109–510). When first produced A1's performance was the same as quoted above for AO. The improvement was found possible without modification by increasing flow rates.

**109-509 A2 Rocket Engine (R II 211 series).**
(As A1 except as follows.)

SEA LEVEL MAXIMUM PERFORMANCE (average figures—*see* end note).

Thrust: 4,410 Lb. Specific impulse: 188 Lb.sec./lb.
Propellant flow rate: 23·4 lb./sec. Effective exhaust velocity: 6,060 ft./sec.
Specific propellant consumption: 19·1 lb./hr./Lb.
(Performance figures above include 1·2 lb./sec. hydrogen peroxide flow to drive turbine.)

DESCRIPTION

*Combustion chamber.*—Operating pressure 368 Lb./in.$^2$ with combustion specific impulse of 198 Lb.sec./lb. at maximum.

*Combustion.*—Minimum 550 Lb. thrust at 74 Lb./in.$^2$ with overall specific impulse of 95 Lb.sec./lb. A steam tapping introduced to permit blowing through chamber to remove liquids at starting, also heat probably improved ignition.

*Starting.*—Electric starter replaced with gravity feed system. Peroxide flow direct into Walter *steam pot* from small container which refills during firing.

*Weight.*—390 Lb. for complete power plant.

*Operational details.*—Developed from A1. Just going into production at the end of hostilities. Air and vapour locks in propellant lines and pumps, experienced in A1 largely prevented by fitting steam ejectors at danger points. Handling and performance much improved. The first A2's were produced giving a performance the same as quoted above for the A1. The improvement was found possible by increasing flow rates.

**109-509 B1 Rocket Engine.** Main engine for Me 163 B2.
(As A1, but with added cruising chamber.)

SEA LEVEL MAXIMUM PERFORMANCE, both chambers (average figures—*see* end note).

Thrust: 3,750 Lb. plus 660 Lb. Specific impulse: 183 Lb.sec./lb.
Propellant flow rate: 24·2 lb./sec. Effective exhaust velocity: 5,880 ft./sec.
Specific propellant consumption: 19·7 lb./hr./Lb.

(Performance figures above include 1·2 lb./sec. hydrogen peroxide flow to drive turbine.)

DESCRIPTION

*Combustion chambers.*—Large: as A1. Small: operating pressure 265 Lb./in.$^2$ with combustion specific impulse of 172 Lb.sec./lb., operated alone: overall specific impulse 155 Lb.sec./lb. at 660 Lb. thrust. Dimensions: maximum diameter 7 in. L* 100 in. Nozzle throat 1·8 in. diameter with 18° expansion half-angle.

*Injectors.*—3 on flat end face. Design as A1.

*Control.*—2 lever system, one per chamber.

*Weight.*—440 Lb. for complete power plant.

*Operational details.*—Unit developed, starting in 1943, to improve upon the cruising efficiency of the A series. It was found to suffer from similar handling troubles as A1 and was dropped in 1944 in favour of 509 C1, which made uses of A2 improvements.

**109-509 C1 Rocket Engine.** Main engine for Ju 263 (also known as Me 163C and Me 263).
(As B1, but with A2 improvements.)

SEA LEVEL MAXIMUM PERFORMANCE, both chambers (average figures—*see* end note).

Thrust: 4,410 Lb. plus 880 Lb. Specific impulse: 186 Lb.sec./lb.
Propellant flow rate: 28·4 lb./sec. Effective exhaust velocity: 5,990 ft./sec.
Specific propellant consumption: 19·3 lb./hr./Lb.
(Performance figures above include 1·4 lb./sec. hydrogen peroxide flow to drive turbine.)

DESCRIPTION

*Flow:*—Turbo-pump speed increased to 17,500 r.p.m. Unit close to limit for efficient pumping at maximum flows.

*Combustion chambers.*—Large: as A2. Small: as B1 design. Operating pressure 309 Lb./in.$^2$ with combustion specific impulse of 184 Lb.sec./lb. Overall specific impulse 165 Lb.sec./lb. when operated alone at 880 Lb. thrust. Small chamber designed to give 60 : 1 expansion ratio at 28,000 ft.

*Injectors.*—Large chamber, as A series, but with slightly larger flow numbers giving required flow with smaller pressure drop.

*Weight.*—420 Lb. for complete power plant.

*Operational details.*—Unit developed in 1944–45 incorporating both A2 improvements and B1 cruising chamber. A modified layout, using same components, gave better pipe runs. Not in production due to cessation of hostilities.

**Note.—Performance figures for the 109–509 series:**

All engines were individually tested, and flow rates were adjusted to give the maximum thrust required for acceptance. Due mainly to the variation in injector-spray characteristics the combustion chamber-flow requirement at a given thrust was found to differ from engine to engine. Flow variation of the order $\pm$ 5 per cent. was considered acceptable. This variation was reflected in flow required to drive the turbine, and also in the overall performance. The figures quoted in the data sheets are the averages obtained for each type during testing and production.

Minimum thrust was limited by the onset of uneven combustion. Considerable variation occurred and was accepted, both in thrust and flow.

The combustion mixture ratio at maximum thrust was not permitted to vary greatly from the mean value, injectors being matched at this flow condition.

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# TECHNICAL REVIEW

Compiled by E. T. B. SMITH, B.Sc., D.C.Ae., A.M.I.Mech.E., A.F.R.Ae.S., FELLOW

The Symposium on High Altitude Rockets and Satellites held at the College of Aeronautics in July will be reported fully in the joint Proceedings to be published later this year, but it seems improper to pass over such a successful gathering without mention. Very briefly, twelve papers were given during the two and a half days of the Symposium, and supporting features consisted of a film show and an exhibition of sounding rockets, items of the Vanguard satellite, and instrumentation.

In addition, a delegate from the U.S.S.R., Dr. Boris Petrov, of the Academy of Sciences, gave a very brief statement on the Russian I.G.Y. Sounding Rocket and Satellite programme. The topics of interest were quite normal, but Dr. Petrov mentioned the use of ejected containers for sampling purposes (as described by Poloskov and Mirtov at the A.E.R.A. meeting in Paris), and also said that the satellites would pass over the Earth in nearly polar orbits, so traversing all the Earth's surface except the central polar regions. No data were given on the size or altitude of the satellites, but the rockets were intended to reach 200 km.

## Progress in Propulsion

Apart from the paper presented to the Joint Symposium referred to above by Dr. L. R. Shepherd, which is a good review of the possible application of nuclear power to space flight, two current notes on fusion reactions may be noted. In *Jet Propulsion*, May, 1957, **27** (5), 570, 572, 574, John Gustavson

lists some of the possible ways in which fusion energy can be utilized. The reactions which may be employed are:

(1) $D + D \rightarrow He^3 + n + 3{\cdot}25$ MeV.
(2) $D + D \rightarrow T + p + 4{\cdot}00$ MeV.
(3) $T + D \rightarrow He^4 + n + 17{\cdot}6$ MeV.
(4) $He^3 + D \rightarrow He^4 + p + 18{\cdot}3$ MeV.
(5) $Li^6 + D \rightarrow 2He^4 + 22{\cdot}4$ MeV.
(6) $Li^7 + p \rightarrow 2He^4 + 17{\cdot}3$ MeV.

Reactions (1) and (2) compete with equal probability, and their end products can then react as (3) and (4), giving high energy yields. The reacting nuclei have positive charges, and the Coulomb barrier must be overcome before fusion takes place: this can be done by supplying sufficient kinetic energy, such as that equivalent to several million degrees Kelvin, at which all the reactants would be in plasma form. To maintain the reaction, the energy release must be greater than the losses, and this can be done by making the reactor big enough, as in the Sun and stars, or by using containing walls which reflect sufficient energy to maintain the reaction. Such walls would have to be made of liquid metals.

Electromagnetic confinement can be employed, if a sufficiently great magnetic field can be applied to produce the "pinch effect"; oscillations in such a pinched discharge may produce spots where fusion takes place efficiently.

A "catalytic" process has been observed in which the negative mu-meson fused deuterons and protons to form $He^3$ nuclei, at room temperature. The short life of mu-mesons at present prohibits this process from being used as an energy source, but perhaps other negative particles might be used.

An unsigned article in *Missiles and Rockets*, June, 1957, **2** (6), 65, gives rather more detail about fusion reactions, saying that the U.K.A.E.A. at Harwell, in a very large glass container, will endeavour to produce a magnetically confined self-sustaining fusion reaction, while they have reached 2,000,000 °K. in small-scale equipment without damage.

Lewis L. Strauss of the U.S.A.E.C. is quoted as saying that, in a magnetically-confined pinch discharge, $10^8$ °K. would be achievable, though initially only for microseconds.

A system which may ultimately utilize power from fusion is that of ion (or plasma) propulsion. Dr. Shepherd in his Cranfield paper mentioned the use of an electromagnetic pump to accelerate plasma, and H. T. Simmons (*Missiles and Rockets*, June, 1957, **2** (6), 76) indicates that much serious attention is being given to the ion-propulsion problem in the U.S.A. The "Snooper" project of Willinski and Orr is for an unmanned ion-propelled vehicle capable of reaching and circling the Moon, as well as Mars and other near planets. It would have an unshielded (fission) reactor, conventional vapour/turbine/generator electricity supply, radiators for waste heat, and ion generators using caesium vapour to give a velocity of 657,000 ft./sec. (but with a thrust of 0·3 lb. wt. for a total weight of 3,500 lb.). The caesium would be kept molten

at about 50 °C. in a pressurized reservoir tank, and passed into a vaporizer at about 800° C. The vapour would then be ionized by hot tungsten grids, and the positive ions would be accelerated by a potential of 27,500 V. The electrons would be introduced to the ion stream beyond the cathode grid through which it passes.

Another article in the same issue of *Missiles and Rockets* (pp. 87–88) by J. E. Naugle of Convair discusses the present state of efficiency of ion systems. The ionization process theoretically needs only 4 to 10 watts per ampere of ions, but in actual systems between 10 and 100 kW. are needed. A "High-Current Ion Ejector" described by W. S. Lamb and E. J. Lofgren (*Rev. sci. Instrum.*, 1956, **27**, 907) produces a 0·75 A. beam of 100 kV. deuterons, with a total power input of 170 kV. The power in the beam is 75 kW., giving a conversion efficiency of 44 per cent. The thrust would be 5,000 dynes (0·01 lb. wt.), with an exhaust velocity of $4{\cdot}4 \times 10^8$ cm./sec. (the plasma beam should be much more efficient, since there is no need to separate ions from electrons: the conducting plasma is pumped like a liquid metal).

Turning to chemical rather than electrical propulsion, we may look at some new "exotic" high-energy fuels, the development of which is partly due to the U.S. Navy "Zip" project (*Jet Propulsion*, June, 1957, **27** (6), 682, 689, 690). These fuels, primarily intended for use in air-breathing engines, are compounds of boron and hydrogen (boranes or borohydrides). When burnt with oxygen, these compounds release about 17,000 C.H.U./lb., compared with 10,000 C.H.U./lb. for hydrocarbons. Data on the available boron hydrides are given in Table I.

TABLE I

| Borane | Formula | Molecular weight | Melting point, °C. | Boiling point, °C. | Specific gravity |
|---|---|---|---|---|---|
| Diborane .. .. | $B_2H_6$ | 27·7 | −165 | −92 | 0·43 |
| Pentaborane .. .. | $B_5H_9$ | 63·2 | −47 | 60 | 0·61 |
| Decaborane .. .. | $B_{10}H_{14}$ | 122·3 | 100 | 213 | 0·94 |

On account of the favourable densities and liquid temperature ranges, pentaborane and decaborane are receiving most attention. Diborane is prepared by adding lithium hydride to boron trifluoride etherate, the diborane coming off as a gas at room temperature:

$$6LiH + 2BF_3(C_2H_5)_2O \rightarrow B_2H_6 + 6LiF + 2(C_2H_5)_2O$$

Plants are being set up in the U.S.A. by Olin Mathieson Chemical Corp. and by Callery Chemical Co.

A. J. Zaehringer (*Missiles and Rockets*, June, 1957, **2** (6), 110, 112) in an article on the chemical aspects of high-energy solid propellents, looks ahead at ideal propellents, which he assumes will be of the composite type (fuel-binder and oxidant) rather than that typified by the cordites. In the organic fuel-binder we should have:

(*a*) High H/C ratio, low total C, some O, some N which can go to $N_2$, no P or S.

(*b*) No metals or halogens (no smoke).

(*c*) Specific gravity of 1 to 2.

(*d*) Low viscosity and vapour pressure, as a liquid. High boiling point, low freezing point.

(*e*) Capability of polymerizing at under 100° C. with fair pot life; no volatile products; no reaction with oxidant; low heat release during cure; short cure time; small shrinkage.

(*f*) Good binding at low (10–25 per cent.) concentrations.

(*g*) Tensile strength over 1,000 lb./in.$^2$: compressive strength over 5,000 lb./in.$^2$.

(*h*) Ability to be cycled over wide temperature ranges, moisture resistance, heat resistance.

(*i*) Cheapness and good availability.

The oxidant should have:

(*a*) Available oxygen content over 50 per cent.

(*b*) No metals or halogens.

(*c*) Specific gravity of 1·5 to 2·5.

(*d*) Insensitiveness to shock.

(*e*) Stability as a solid to 200–350° C.

(*f*) Low coefficient of thermal expansion.

(*g*) Low hygroscopicity.

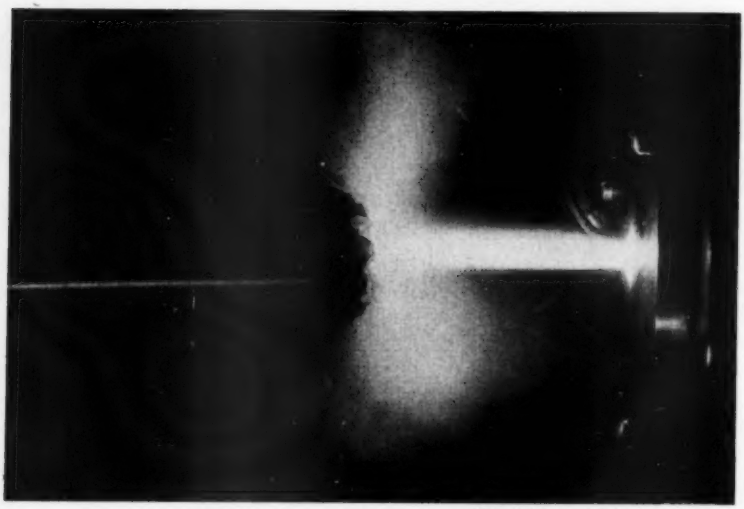
(*h*) Low cost and good availability.

Zaehringer states that solid propellents should be capable of being pushed into "the ultra-energy class," but a contributor to the discussion at Cranfield (Dr. Johnson of the Ministry of Supply) made the point that solid propellents are like high explosives and never properly under control, and that as the energy content is increased so too is the liability to spontaneous detonation.

A. P. Purmal (*Zhur. fiz. Khim.*, 1955, **29**, 846–849) has detected the existence of a hydrogen superperoxide ($H_2O_4$) during the photochemical decomposition of hydrogen peroxide. This should have about three times the available oxygen of hydrogen peroxide, together with higher density, easier catalytic decomposition, and wider liquid temperature range—if it could be made.

**Test Facilities**

Plasma jets or stabilized arcs have been discussed previously in TECHNICAL REVIEW, but some photographs are now available which show the kind of effect which is achieved. Fig. 1 and 2 were taken of the tests on a blunt-nosed steel body in the Giannini Research Laboratory's system, which consists of an electric arc in a water-filled pressure vessel, fitted with a nozzle to direct



[*Giannini Research Laboratory photograph*

FIG. 1. About one second after a 15,000° K. plasma jet meets a solid steel aerodynamic body. The blunt nose is beginning to melt.

[*Giannini Research Laboratory photograph*

FIG. 2. A fraction of a second later, nose is rapidly losing its aerodynamic shape.

the dissociated steam on to the specimen. The maximum temperature is about 15,000° K. Fig. 1 shows the start of melting of the body after about one second; Fig. 2 shows the complete loss of "aerodynamic" shape a fraction of a second later. An illustration in *Aviation Week*, March 25, 1957, **66**(12), p. 31, shows the 70 kW. jet of the General Electric Co., U.S.A.; according to the caption, the temperature is 2,000° F. (1,400° K.), but it is unlikely that, at normal atmospheric pressure, much plasma can exist at this temperature, unless the rate of re-association of the water vapour is actually much slower than it is believed to be. The Company proposes to build a 10,000 kW. plasma jet, using a 3 in. hollow cathode instead of $\frac{1}{4}$ in. as in the present unit.

A list, showing the range of conditions which can be achieved in hypersonic test facilities in the United States, has been published (*Jet Propulsion*, May, 1957, **27** (5), 548) by the General Electric Co., U.S.A., and is given in Table II.

The U.S. Naval Research Laboratory reports (*Aviation Week*, March 25, 1957, **66** (12), 26) the attainment of velocities of 450,000 ft./sec., at temperatures of several hundred thousand degrees Kelvin, for several tenths of micro-seconds. Such velocities represent Mach Numbers of 200, and they have been achieved by the electromagnetic acceleration of ionized gas (hydrogen, deuterium, helium). A quartz tube a few centimetres in diameter, containing the gas, is located perpendicular to an intense magnetic field; electrodes in the tube allow a current to be passed through the gas, the resulting force on the gas producing a luminous shock front which travels with the velocities described. The velocity is measured either by photographing the front with a moving-film camera or by using the Doppler shift in 9 mm. microwave radiation reflected from the front: the temperature is computed from the velocity, and checked spectroscopically.

The apparatus for producing the shock waves is fairly cheap and simple, the power supplies coming from banks of capacitors; 125,000 V. and 300,000–400,000 A. are used, for periods of up to one microsecond, but the instrumentation costs are very great.

**Vehicle Progress**

Some information has been given about the performance and characteristics of the U.S. vehicles X-2, X-15 and X-17, sponsored by the N.A.C.A. X-2 (*Aviation Week*, July 1, 1957, **66** (26), 52), which is a manned rocket aeroplane made by Bell, reached an altitude of 126,200 ft. in 1956. X-15, which is a manned rocket aeroplane expressly designed (by North American Aviation) to attain extreme altitudes and speeds, is due to fly in 1958, should attain a Mach number of 6 to 7 at 100 miles altitude. It will be built of temperature-resistant materials such as Inconel-X, since stagnation temperatures of between 800° and 1,600° C. may be encountered. Insulation and cooling for the pilot's compartment will be vital, as will be the escape system, using a special survival suit. The liquid propellant rocket is manufactured by Reaction Motors Incorporated, and will have more thrust than the motor in the X-2.

An X-17 hypersonic unmanned test vehicle, built by Lockheed, has inadvertently gone higher and further than it was intended to (*Aviation Week*, April 22, 1957, **66** (16), 31). Instead of the X-17 tilting and continuing to

TABLE II

| | Velocity ft./sec. | Mach No. | Stagnation pressure ratio | Stagnation temperature ratio | $\dot{q}$ B.T.U./ ft.²/sec. | Model diameter, ft. | Flow duration, sec. | Medium |
|---|---|---|---|---|---|---|---|---|
| *Intermittent Wind Tunnels—* | | | | | | | | |
| Air | 7,000 | 8 | 0·3 | 6 | 100 | 1 | 15 | Air |
| Helium | 6,000 | 30 | 0·01 | 4 | 30 | 0·1 | 15 | He |
| Electrical | 17,000 | 15 | 2·5 | 21 | 300 | 0·9 | 0·1 | Air, C |
| *Mass Accelerator—* | | | | | | | | |
| High explosive driver | 2,500 | 23 | 70 | 32 | 5,000 | 0·7 | 0·001 | Air |
| Combustion driver | 16,500 | 15 | 20 | 21 | 2,500 | 0·05 | 0·01 | Air |
| Electrical driver | 25,000 | 23 | 70 | 32 | 5,000 | 0·1 | 0·01 | Air |
| *Shock Tube—* | | | | | | | | |
| High explosive driver | 20,000 | 3 | 900 | 45 | 16,000 | 0·2 | 0·0001 | Air |
| Combustion driver | 15,000 | 3 | 300 | 35 | 8,000 | 0·2 | 0·0005 | Air |
| *Shock Tunnel—* | | | | | | | | |
| High explosive driver | 20,000 | 20 | 1·0 | 37 | 1,000 | 1 | 0·0005 | Air |
| Combustion driver | 15,000 | 15 | 0·5 | 33 | 500 | 1 | 0·002 | Air |
| Stabilized arc | 10,000 | 3 | 1 | 35 | 4,000 | 0·5 | 30 | H, O, C |
| Rocket exhaust | 8,000 | 3 | 5 | 10 | 1,500 | 2 | 30 | Combustion products |
| Solar furnace | 0 | 0 | 1 | 10 | 2,000 | 0·02 | 60 | Air |
| Plasma accelerator | 250,000 | 250 | 20 | 3,300 | — | — | $10^{-7}$ | H, He, Air |

The stagnation pressure and temperature ratios refer to ambient conditions as standard.
$\dot{q}$ is the maximum heat transfer rate on the surface of the specimen.

be driven downwards to fulfil its normal function as a re-entry test vehicle, the guidance failed to act, and a range of 700 miles and height of 600 miles in a ballistic trajectory was achieved instead of only the normal range of about 200 miles. Out of about twenty vehicles fired, seventeen have functioned satisfactorily, simulating I.C.B.M. re-entry (at a greater altitude) by means of the downward-directed thrust: speeds of 9,000 m.p.h. have been achieved. Leading details of X-2, X-15 and X-17 were given briefly in a recent TECHNICAL REVIEW (*J.B.I.S.*, 1957, **16**, 166).

The state of affairs in mid-July regarding the U.S.A.F. ballistic missiles (*Aviation Week*, July 15, 1957, **67** (2), 29) was that Thor, the I.R.B.M. built by Douglas, had been fired three times: the first vehicle had a propellent valve misbehave immediately after lifting from the ground, and blew up: the second tilted after take-off and was destroyed by the range safety officer: and the third was destroyed when being refuelled, due to failure of a fuel pressure regulator.

The first and only launching* of the Convair Atlas I.C.B.M. was unsuccessful due to the engine gimbal system failing, after a total operating time of 32 seconds (22 on the ground, 10 in the air). Previously, 700 seconds burning time had been amassed. Perhaps that is why the system failed when it was fired in earnest (*Missiles and Rockets*, July, 1957, **2** (7), 46, says that Atlas No. 1 failed because of a faulty booster engine turbo pump). The vehicle has two boost engines and a sustainer.

The U.S. Army's "Satellite Vehicle," Jupiter "C," is discussed in *Missiles and Rockets*, July, 1957, **2** (7), 44. Jupiter itself is an I.R.B.M. development of the Redstone vehicle, and Jupiter "C" consists of a Redstone used as a first stage with two stages of clustered solid propellent rockets, with a small payload consisting of a transmitter and some instrumentation. The first firing of a Jupiter "C" in September, 1956, is supposed to have given a range of 3,300 miles: it was fired from Patrick Air Force Base, Florida, and the transmitter was tracked all the way. The final part of the powered trajectory was depressed so that the payload came back to earth, but with suitable direction the payload could have been put into an orbit, and it is suggested that the second Jupiter "C" may in fact do this. The first satellite project considered in the U.S.A. ("Project Orbiter") was to have been based on Redstone with a cluster of Loki rockets as the second stage, but was cancelled when Vanguard was introduced. The satellite tracking system developed by the U.S. Army is called Microlock, and was developed by Jet Propulsion Laboratory, who also had a hand in developing the vehicle. The satellite transmitter weighs two pounds and has a useful life of three months, and provides two narrow-band telemetry channels: the system is supposed to track successfully over a range of 3,000 miles at any azimuth or elevation angle.

The first U.S. rocket firings of the I.G.Y. were made early in July from Pt. Mugu Naval Air Missile Test Centre, using DAN rockets, consisting of Deacon solid propellent motors as sustainers with Nike-Ajax boosts (*Aviation Week*, July 15, 1957, **67** (2), 65). The Naval Research Laboratory are using the rockets in an attempt to correlate ultra-violet, X-ray and gamma-ray

* At the time of writing.

emissions from solar flares with deep radio fading due to increases in electron density in the atmosphere. The first launching was designed to measure normal "quiet" background signals, for comparison with data obtained during flares. Investigations during flares need very rapid response on the part of the firing crews and observers, and the DAN rockets can be fired at one minute's notice. Astronomical observatories at Climax, Colorado, Sacramento Peak, New Mexico and Mount Wilson, California, will observe the Sun and notify the firing site by teleprinter or radio, so that the rocket should be at altitude four minutes after a flare being seen to begin: flares may reach peak intensity in five minutes. DAN is capable of altitudes of 80 miles, more than adequate to cover the 50 to 70 mile altitude regions (D and E layers) where much atmospheric ionization takes place.

The results of the firings showed that there was no X-radiation and a lower ultra-violet level than had been anticipated.

A new balloon-launched rocket system is that called "Farside," sponsored by the U.S.A.F. Air Research Development Command (*Aviation Week*, July 22, 1957, **67** (3), 29) which will attempt to get a payload to a distance of 4,000 miles from the Earth (an Earth radius away). Six rounds are being built, and will be launched from Eniwetok Atoll in the Pacific Ocean. The balloon will be 200 ft. in diameter, made of polyethylene film and filled with helium: it weighs 1,500 lb., and should rise to 100,000 ft.—above 90 per cent. of the Earth's atmosphere. The rocket itself will have a payload of 10 lb., and will be 30 ft. long, the base diameter being 8 ft., and the diameter over the last stage will be 4 in. The structure is largely of aluminium alloy, with stainless steel for the nose cone and payload section. Four stages of propulsion will be used, all solid propellent motors: first stage comprises four Thiokol Recruits, as used to boost the X-17, second stage one Recruit, third consists of four ASP (Grand Central Rocket Co.) motors, and the fourth is a single ASP motor. The total burning time will be 8 seconds, and the maximum velocity will be 25,000 ft./sec. The first two stages will fall in the sea, but the other two will burn up on re-entry into the atmosphere. Instrumentation is designed to withstand the extremely high accelerations, and will gain data on magnetic and gravitational fields, cosmic rays, ionospheric properties, and meteorite densities.

Some details of the Vanguard Satellite were given by R. C. Baumann of Naval Research Laboratory at the A.R.S. Spring Meeting (*Jet Propulsion*, April, 1957, **27** (4), 444, 446). He said:

> "It is a sphere, 20 in. dia. made of FS1 magnesium alloy with a highly polished coat of silicon monoxide. Coming off the sphere at the equator are four antennas mounted 90° apart. Tubular rods fasten the antennas to a tubular ring which is concentric with a cylindrical inner package.
>
> "Four bow-shaped, vertical, tubular members, spaced 90° apart and at 45° to the antenna supports, also brace the concentric tubular ring. The four bows, in turn, are fastened to the support ring of the access port at the top, and to the main support column, which houses the separation mechanism, at the bottom.
>
> "The cylindrical internal package is mounted on the main support column and is secured to the concentric tubular ring by four low thermal conductivity radial supports, spaced 90° apart and angularly positioned between the antenna supports and the vertical members.

"Inside the package are five ¾ in. thick modules: one for the Minitrack transmitter and associated electronic equipment, one for the Lyman alpha electronics and batteries, one for the coding 48-channel telemetering system, one for the peak memory orbital switch unit, and one for the meteor counter. Below these five are two more modules for battery packages. All seven modules are fastened to the top cover of the internal package by two ¼ in. dia. rods.

"On the shell are four microphones, a Lyman alpha solar cell, and a Lyman alpha ion chamber. In addition, there are erosion gauges and temperature gauges which are located both at the north pole area and around the equator. Then there is a pressure gauge which is located on the vertical axis underneath the main package, and two pressure zones or bands that girdle the sphere above and below the equator."

*Missiles and Rockets*, July, 1957, **2** (7), 118, lists the investigations which will be performed in the four satellites which, it is stated, it is hoped to establish in orbits with the six "business" firings, as distinct from seven test firings. These investigations are, in order of firing:

(1) NAVAL RESEARCH LABORATORY: Ultra-violet, temperature and meteor measurements.

(2) STATE UNIVERSITY OF IOWA: Cosmic rays; U.S. AIR FORCE: Meteor measurements.

(3) NAVAL RESEARCH LABORATORY: Magnetometer; N.A.C.A.: Expandable plastic foil sphere.

(4) *either* UNIVERSITY OF WISCONSIN: Energy balance; *or*
U.S. ARMY SIGNAL CORPS: Cloud distribution measurements.

**Electronic Devices**

Development in the solid-state electronic field is discussed in three reports (*Aviation Week*, April 8, 1957, **66** (14), 86; May 13, 1957, **66** (19), 101; July 22, 1957, **67** (3), 33). The possibility is forecast of making complete amplifiers as aggregates of semiconductor p-n junctions, which would function as a conventional triode preamplifier and cathode-coupled amplifier. All resistors except one would consist of back-biased junctions, and power could be obtained by light falling on three of the eleven junctions in the amplifier.

The initial step towards such complicated circuit systems would appear to have been made in devices which operate similarly to multi-element thermionic valves: the General Electric Co., U.S.A., has developed a "dual-triode" device, called a "tetra-junction" transistor, which has n-p-n-p-n junctions.

Other new solid-state devices include the Cryotron, which depends on the control which can be exerted over the superconductivity of a metal, at liquid helium temperatures, by the application of an external magnetic field. This can at present be used only as a switching device, but there is some likelihood of it being adapted to act as a triode amplifier. Another is the Hall gyrator: this is a ferrite device using the Hall effect, which, used at high frequencies, can act as a radar isolator, but may also have applications at low frequencies. A device, similar to the cryotron but working at room temperature, uses indium antimonide: the conductivity of this compound varies rapidly with the applied magnetic field, and so can be used as a rheostat, an adjustable voltage divider, or a relay, all without moving parts.

A special multi-element transistor can be used as a commutator to switch an input to any one of several outputs: switching times of 0·5 microsecond are achieved, with power capacity of 50 W., and isolation of 20 dB. between adjacent circuits. A luminescence-effect amplifier has been made which converts an alternating current input to a luminescence signal, controlling an amplified d.c. output, and it is reported to have power gains of up to 70,000. A microwave modulator has been constructed by using a thin slice of germanium in a waveguide and, by applying a direct voltage or external field, the microwave absorption of the germanium is altered, at rates up to 100/sec. with pulse widths of one microsecond. Another device makes use of photoconductive switching: a square matrix, with photoconducting material at the intersections of the rows and columns, could be switched by focussing a point of light at the desired intersection, using a cathode ray tube close to the matrix panel.

Semiconductor junctions can also be used as capacitors, and where a reverse bias is used high capacitance per unit area is obtained with very high "Q." Some junction diodes have voltage-dependent capacitances, which make them useful for such applications as automatic frequency control instead of reactance valves.

Conventional germanium junction transistors have cut-off frequencies which are often no more than 100 Mc./sec.: larger power transistors may in fact be limited to the "audio" range, up to, say, 1 Mc./sec. In addition, transistors suffer from low input impedance, high output capacitance, and coupling between input and output circuits. Many of these troubles are due to the relatively low speed of motion of electrons and holes through the transistor base, and a device called a "Spacistor" has been developed by Raytheon Manufacturing Co. The device is claimed to be able to produce power gains of 70 dB. and voltage gains of 3,000, with an upper frequency limit for amplification of 10,000 Mc./sec. and it consists of a single p-n diode junction, which eliminates the conventional transistor base. A reverse-polarity voltage source and the output load are connected across the diode, the reverse bias producing a space charge region at the junction. An element called an injector is the source of load current, and is biased at a potential lower than that of the space charge region to which it is connected, so that it injects electrons: the strong field causes the electrons to move rapidly to the collector and load. A fourth electrode called a modulator is connected to the space charge region through a p-type junction and biased to zero current: it produces a field in the space charge region whose strength depends on the modulator voltage. An input signal applied to the modulator changes the potential in the space charge region, so controlling the flow of electrons out of the injector and into the load via the collector. The modulator also prevents feedback between output and input: it effectively anchors the potential of the space charge region between the modulator and the base, so isolating the load current from the injector.

In general, solid-state devices show enormous savings over thermionic valve circuits in both size and power consumption, but they now have rivals, at least in some fields: devices called "Solions" (ions in solution) may become

available as transducers and amplifiers (*Aviation Week*, July 1, 1957, **66** (26), 37). They depend on the motion of ions in liquid solutions, and, in the case of a typical integrator, an aqueous solution of potassium iodide, with a small added quantity of iodine, is placed in a chamber with a porous dividing wall. Two electrodes are placed in one compartment, and a small bias is applied between them. As signals are applied to the electrodes, the iodine is electrolysed to form more potassium iodide, which penetrates the porous wall and so alters the iodine concentration in the electrode compartment: the current flowing is then a measure of the total iodine conversion and hence the integrated signal.

**Tektites**

Tektites are defined (*Chambers's 20th Century Dictionary*, New Mid-Century Version) as "glassy (supposed) meteorites." Their true origin is discussed in a paper in *Nature, Lond.*, 1957, **179**, 556, in which Prof. H. C. Urey of the University of Chicago presents a new hypothesis. Tektites are small, glassy pebbles found widely scattered over the Earth, for example, over nearly all of southern Australia. These particles show flow marks on their surfaces compatible with passage through an atmosphere at very high velocity, and they are similar to sedimentary rocks without their water content: they contain several per cent. of potassium oxide, large proportions of silica and alumina, and 1·5 p.p.m. of uranium. The chemical composition is only produced on Earth by complicated processes, such as the weathering of granite and basalt and the removal of the alkaline earths and iron by the action of water, and volcanic sources of heat are not sufficient to cause melting. If they were of extraterrestrial origin, either as a compact mass or a swarm of particles, the tektites would not have been distributed over the Earth in the ways observed. For instance, a large mass would have given a distribution over only a few kilometres, and a shower, if dense enough to avoid being broken up completely by solar gravitation, would have covered southern Australia to a surface density of 100 g./cm.$^2$, which is not the case. Meteor impact would have left a crater of some sort. Prof. Urey now proposes that the Earth collided with a comet. The velocity of a comet in a parabolic orbit, at the Earth's distance from the Sun, is 42 km./sec., and with a diameter of 10 km. and density of 0·01 gm./cm.$^3$, the kinetic energy of such a spherical comet would be $5 \times 10^{28}$ ergs (half a million hydrogen bombs).

Entry of the atmosphere by such an object would result in heating and compression of the material, conversion to gas taking place, and the silicates would be volatilized as fine dust. The explosive reaction would begin at 60–100 km. above the Earth, and the high-temperature mass would move to the surface, melt it, and, with the expansion of the highly compressed gas, blow terrestrial material in all directions at high velocity. The process would take about one second, and the pressures would be about 40,000 atm. Such a process is held to account for many of the puzzling features of tektites, including their varying composition (from the Earth's surface), the high temperature required for melting and the retention of alkali by rapid melting and cooling,

the observed wide scattering, and the flow marks (perhaps made by gas blowing over stationary particles).

The number of comets arriving near the Sun is about ten per year, and it is estimated that about one collision with the Earth should occur in about fifty million years: the numerous groups of tektites which are observed may be due to cometary heads breaking up before striking the Earth simultaneously at different points. Many tektites are Pleistocene in age (i.e., contemporaneous), but some in Texas are Eocene (i.e., about fifty million years old).

**High Temperatures and Resistant Materials**

Although we know that high temperatures are encountered during re-entry, and in rocket combustion chambers, it is not easy to obtain simple methods of producing these temperatures on a laboratory scale. One promising device in this field is the solar furnace, which is fairly cheap to make and costs nothing to drive: all that is needed is a clear sky, which perhaps is why the method is not widely used in the United Kingdom.

N. K. Hiester, T. E. Tietz and E. Loh (*Jet Propulsion*, May, 1957, **27** (5), 507, 513, 546) describe work at the California Institute of Technology on the theoretical performance of solar furnaces. They show that while, with a rim angle of the reflecting paraboloid of 90°, it is theoretically possible to obtain a temperature at the focus equal to that of the photosphere, i.e., 5,800° K., this temperature is reduced by imperfections in the reflectivity and figuring of the mirror, losses from the specimen, and absorption of radiation in the atmosphere, and it is suggested that 4,000° K. should be obtained, even though no reliable report of such temperatures are available. The heat flux, based on a solar constant of 2 cal./cm.$^2$/min. in the stratosphere, may approach $30 \times 10^3$ cal./cm.$^2$/min. The solar furnace at the Institute uses an array of lenses, each 2 ft. in diameter, to concentrate the sunlight: this has a maximum attainable temperature of 4,700° K., with a cavity receiver.

An article on high temperature materials (*Jet Propulsion*, June, 1957, **27** (6), 644–649) by J. R. Kattus describes tests done at temperatures up to 2,870° C. (or melting points) on copper, iron, molybdenum, tantalum and graphite, to determine their short-time mechanical properties for use with kinetically heated vehicles. Rupture was caused in either about one second or thirty minutes, depending on the loading rate, and the heating was provided by passing a heavy current through the specimens, which were exposed to the air. The metals tantalum and molybdenum, while stronger at high temperatures than ingot iron or copper, oxidized very rapidly, and would not be satisfactory for service of more than a few seconds duration above, say, 2,200° C. Graphite, though brittle, can be used up to, say, 2,500° C. without much oxidation. For lower temperatures, its very high thermal conductivity gives copper significant advantages.

A new non-metallic material with good refractory properties is called Pyroceram (*Jet Propulsion*, June, 1957, **27** (6), 690). This has been developed by the Corning Glass Works in Pittsburgh, and is basically a hard glass which has added to it nucleating agents of an unstated nature. The glass can be worked by conventional means, and then, by cooling and subsequent heat

treatment, the nucleating agents form sub-microscopic crystals with dimensions of up to 1,000 Ångström units. After final cooling, the material can be machined and can be obtained either glassy or opaque, and is as impervious to gases as good quality glass.

Thermal expension can be controlled to give coefficients in the range from slightly negative to about $200 \times 10^{-7}$/°C., which will match heavy metals. Specific gravity is about 2·50, and the softening temperature is at up to 1,350° C. The strength is high, up to 60,000 lb./in.$^2$ in flexure, so that the strength-weight ratio is excellent. The material should be cheaper than stainless steel.

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## ERRATA

The following corrections should be made to TECHNICAL REVIEW, *J.B.I.S.*, April-June, 1957, **16** (2), 100–108.

Page 106, line 8 from bottom. *For* **65** (3), *read* **66** (3).

Page 106, line 3 from bottom. *For* 11,0003 F' (60903 C.), *read* 11,000° F. (6090° C.).

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# CORRESPONDENCE

(In these letters to the Editor, some passages have necessarily been omitted).

### Pirouettes in Spaceships

Following up Mr. Lawden's article[1] on the simulation of gravity, it is entertaining to examine the effect of trying to turn round on a spaceship or spacestation that is itself rotating. For this purpose, we represent a man by a solid cylinder of length $l$ ft., radius $a$ ft., and mass $M$ lb. If we consider his axis along a radius of the spaceship or spacestation (see Fig. 1) with his head pointing towards the centre of rotation, then if he remains stationary the only force he is aware of is the centrifugal acceleration acting from head to foot and thereby simulating gravity.

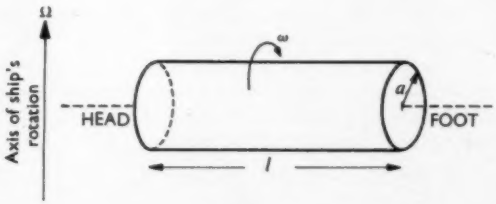


FIG. 1.

But if now he turns (about his own axis) the various parts of his body will have velocity relative to the ship and will therefore become subject to Coriolis "force." By setting our cylindrical man spinning at a rate of $\omega$ radians/sec. we can easily calculate the nett effect of Coriolis force on the body. If $\Omega$ radians/sec. is the angular velocity of the ship (or station) then we find we get a resultant couple of moment

$$C = \tfrac{1}{64}\, a^2\, \Omega\, \omega M \text{ lb. wt. ft.}$$

tending to topple the man over in such a way that his head would tend to move in the opposite direction to that shown by the vector $\Omega$ in the sketch. The man is in fact behaving like a gyroscope under the influence of a precession torque due to the ship's rotation.

As a numerical example we take an extreme case. Suppose our cylindrical man were on a space ship of 10 ft. radius which is being whirled round to give 32 ft./sec.$^2$ (*i.e.*, one Earth "$g$") at the circumference. Then $\Omega = 1{\cdot}8$ radians/sec. Suppose our man is short and fat (height 5 ft., radius 1 ft.) and weighs 180 pounds. Now suppose he twists round at an angular velocity of 2 radians/sec. (*i.e.*, one complete turn in about 3 seconds) then we find the Coriolis couple is $C = 10$ lb.wt.ft. This is equivalent to a force of 4 lb.wt. pushing him in the centre of gravity.

This does not appear to be excessive, and under more serene circumstances the toppling force will only be of the order of 1 lb. wt. There is the point that this force remains fixed in direction in space and therefore tends to push one from different apparent directions as one turns round, but it seems unlikely to be a serious inconvenience. A quick calculation shows that our short, fat cylindrical man need lean over only $1\frac{1}{2}°$ to balance the toppling force.

86 Marston Gardens, Luton. D. J. CASHMORE.

*September* 25.

REFERENCE

(1) D. F. Lawden, *J.B.I.S.*, 1957, **16**, 134.

## Meteoric Erosion

SIR,

In TECHNICAL REVIEW, *J.B.I.S.*, **1957**, **16**, 103, an estimate of meteoric sandblasting rate is presented. The estimated rate of 10 microns loss in 1,600 seconds seems to me to be far too high. We have some observational evidence in that meteoric dust should erode meteors as well as spaceships. At this rate, a spherical meteor one centimeter in diameter would be reduced to one half that diameter in 400,000 seconds. Also at that rate, a moon the size of Phobos would be reduced to a peanut in $1{\cdot}6 \times 10^{12}$ seconds. Since meteors and moons do exist, this estimate of erosion rate appears much too high.

264 N. Pleasant Street, Amherst, Massachusetts, U.S.A. ED MIKSCH.

*August* 1.

# B.I.S. NEWS

## Council

As briefly announced on p. 181 of the July-September issue of the *Journal*, the following members of Council were elected with effect from 3 August, 1957: Dr. L. R. Shepherd, D. J. Cashmore, D. Hurden, and E. T. B. Smith. The new Council met for the first time on 17 August, 1957 and elected Dr. L. R. Shepherd as Chairman of Council, and Mr. K. W. Gatland as Vice-Chairman.

At this meeting it was also learnt that the Secretary had received the resignation from the Council of Mr. R. A. Smith, who was one of the earliest members of the Society, has served on the Council without a break since the Society was reformed at the end of the war, and was Vice-Chairman 1953-56 and Chairman 1956–57. No doubt members will wish to join in thanking Mr. Smith for all his work to date on behalf of the Society and astronautics in general.

The following standing committees were appointed:

*Finance and General Purposes*: L. J. Carter, K. W. Gatland, Dr. L. R. Shepherd, G. V. E. Thompson.

*Membership*: K. W. Gatland, D. Hurden, G. V. E. Thompson.

*International Astronautical Federation*: L. J. Carter, K. W. Gatland, Dr. L. R. Shepherd.

*Journal Publications Committee*: D. J. Cashmore, Dr. N. H. Langton, L. R. Shepherd, A. E. Slater, G. V. E. Thompson.

*Spaceflight Editorial Board*: A. Coleman, C. A. Cross, D. Hurden, Patrick Moore, A. E. Slater.

It was agreed that K. W. Gatland and L. J. Carter should act as B.I.S. delegates to the Barcelona Congress.

At the Council meeting held on September 14, matters to be discussed at the business meetings of the Congress were considered. It was decided to co-opt Dr. N. H. Langton to fill the Council vacancy caused by the resignation of Mr R. A. Smith. A long report submitted by the Finance and General Purposes Committee concerning the future financial policy of the Society was adopted; some of the recommendations approved are described in detail below.

At its meeting on September 28, two *ad hoc* committees were set up:

*Technical Policy Committee* (to consider ways of enhancing the technical status of the Society): L. J. Carter, D. J. Cashmore, S. W. Greenwood, W. N. Neat, Dr. L. R. Shepherd, E. T. B. Smith, G. V. E. Thompson.

*25th Anniversary Celebrations Committee*: L. J. Carter, K. W. Gatland, A. M. Kunesch, Dr. N. H. Langton.

The Secretary will be glad to receive suggestions concerning the form the twenty-fifth anniversary celebrations should take.

## Membership Fees

*Spaceflight* has now been appearing for over a year and has been widely welcomed. Its publication has meant that members have been receiving eight issues per year (four of *Spaceflight* and four of the *Journal*) instead of the previous six copies of the *Journal* per annum, without any increase in subscription. This has resulted in a deficit for the year 1956/57, and with the recent

severe increases in postage rates, both inland and foreign, the position is likely to deteriorate still further. Some increase in subscription is obviously necessary; indeed, in these inflationary times it is only by very careful management of the Society's finances that it has been found possible to hold the subscriptions at their present level for some years. Council has decided to recommend to members that the following scale of fees be adopted:

| | |
|---|---|
| Fellowship and Senior Membership | £3 3s. 0d. |
| Ordinary Membership | £2 2s. 0d. |
| Student and Junior Membership | £1 11s. 6d. |

A special circular letter from the Chairman explaining fully the reasons for this recommendation has been sent to members together with a voting paper. If the recommendation is approved, the new fees will come into operation as from 1 January, 1958.

For similar reasons, it has also been found necessary to increase the face value of the *Journal* from 4s. to 5s. per copy, and that of *Spaceflight* from 3s. to 3s. 6d. Library subscriptions are also being increased.

### Compounded Subscription Rates

Consequent upon the increase in membership fees, new compounded subscription rates are being introduced as from 1 January, 1958, and are as follows (present rates in parentheses):

| Age | Membership | Fellowship and Senior Membership |
|---|---|---|
| 20–25 | 41 (36) | 50 (45) |
| 26–30 | 37 (33) | 45 (40) |
| 31–35 | 34 (30) | 40 (36) |
| 36–40 | 31 (27) | 36 (32) |
| 41–45 | 28 (24) | 33 (29) |
| 46–50 | 26 (22) | 30 (26) |
| 51–55 | 23 (20) | 26 (23) |
| 55–60 | 21 (18) | 24 (21) |
| over 60 | 18 (18) | 21 (21) |

It is hoped that as many members as possible will take advantage of this method of payment. The Society benefits by having the immediate use of a lump sum (these payments are invested and credited to a special account in readiness for future use in purchasing headquarters for the Society or similar purposes); the member concerned benefits by paying a much smaller sum than would be required in separate annual fees, particularly in an inflationary period. Until December 31, 1957, subscriptions may be compounded at the present rates.

### Publications: Binding Arrangements

The index for Volume 15 of the *Journal* was sent to members with the July-September issue. Members who wish to have them bound should send them, together with the index and a remittance of 11s. 0d. per volume, direct to

W. Heffer & Sons, Ltd., Hills Road, Cambridge. Please do not send them to the Society.

As only four issues of the *Journal* now appear per annum, it has been decided that Volume 16 of the *Journal* will run until the end of 1958, to provide sufficient bulk (eight issues) for binding. Members are asked to note therefore, that no index will be issued until the end of this period; copies should not be sent for binding until the volume is complete. A similar arrangement is being made for *Spaceflight*, Volume 1 of which will run until the end of 1958, and is thus scheduled to contain nine issues.

It is hoped that the new subscription rates referred to above will not only enable the Society to wipe out the present deficit, but also permit some further improvement in the Society's publications at the end of 1958, when we may be able to introduce a better format for the *Journal* and return to bimonthly issues, without any reduction in the number of issues of *Spaceflight*. This is particularly desirable now that more suitable material is being received for publication and may be expected to be available following the successful launchings of the Russian satellites. With a quarterly publication delays are caused both by the shortage of space and the long gaps between successive issues. Members are asked to do all they can to reduce the cost of the Society's publications by encouraging their firms to advertise in both the *Journal* and *Spaceflight*.

### Students Meeting

A meeting has been arranged in the Grand Hall, Caxton Hall, London, S.W.1, on Saturday, December 28 at 2.30 p.m. for students and scholars in the sixth forms of public, grammar and secondary modern schools in the London area. This is not intended to be part of the Society's ordinary lecture programme, but the presence of a few members prepared to act as stewards is desirable, and volunteers are asked to write to the Secretary.

### Subscriptions to "Sky and Telescope"

Arrangements have now been made for the Interplanetary Publishing Co. to accept subscriptions for *Sky & Telescope*, copies of which will be mailed from America immediately on publication to save time. This well-illustrated monthly publication is a "must" for anyone interested in astronomy; in addition to star charts, notes on telescope making, and many excellent feature articles, it gives the most extensive coverage on astronomical news available to the English-speaking world.

The subscription rates are as follows:

| | |
|---|---|
| One year | £2 10s. 0d. |
| Two years | £4 12s. 0d. |
| Three Years | £6 15s. 0d. |

As far as possible, subscriptions will be deemed to relate from the preceding November 1 (which is the date when each new volume begins), and an annual subscription will therefore always cover a complete volume. Single copies may be ordered at 4s. each.

Remittances and enquiries should be sent to the Interplanetary Publishing Co., 12 Bessborough Gardens, London S.W.1.

## Election of Members

The following elections were made at the Council Meeting of 18 May, 1957:

*Fellows.*

BEN GEORGE BROMBERG, Sc.D., 811, Jackson Street, St. Charles, Missouri, U.S.A.
NATHAN NORMAN BUDISH, B.Ae.E., A.F.R.Ae.S., Laurel Avenue, Kingston, New Jersey, U.S.A.
DONALD CANT, M.B.B.S., F.C.C.P., 23, Farm Road, Corner Brook, Newfoundland, Canada.
EDWARD H. CARTER, B.S., 8201, Aura Ave., Reseda, Calif., U.S.A.
BRYAN SANDFORD CLARK, 26, rue de la Confederation, Geneva, Switzerland.
BRIAN PATRICK CLEAR, 4, Eden Avenue, Culcheth, Warrington, Lancashire.
JOHN GRIMSON, G.I.Mech.E., Lanchester Hall, College of Aeronautics, Cranfield, Bletchley, Bucks.
ROBERT THOMPSON HOLMES, B.Sc., 32, Caerlaverock Avenue, Prestwick, Ayreshire.
JOHN VAUGHAN BRUCE JERVIS-READ, B.A., 63, Barkston Gardens, London, S.W.5.
DEVEREUX LLOYD KAVANAGH, M.S., RD No. 3, Bellefonte, Penn., U.S.A.
CARL EDWARD SAGAN, A.B., M.S., Yerkes Observatory, Williams Bay, Wisconsin, U.S.A.
FRANK ROLLINGS SMITH, B.Sc., 13, The Crescent, Belmont, Surrey.
ERIC JOHN THOMPSON, B.Sc., F.S.S., 15, Arthur Road, Holloway, London, N.7.
FREDERIC VAN VYVE, Fabrique Nationale D'Armes de Guerre (F.N.), Herstal, Belgium.

*Members.*

Colonel VLADIMIR AFANASIEV, 16, Kensington Palace Gardens, London, W.8.
FRANKLIN JOHN ALLEN, 11, Orchard Road, Birstall, Leicester.
DOMINGOS BAHI, Rua Correa Dias, 161, Vila Mariana, Sao Paulo, Brasil.
DON A. BAKER, B.S., OMS 290, Kirtland AFB, New Mexico, U.S.A.
JAMES NELSON BENFORD, Liliencronstrasse 10, Frankfurt/Main, Germany.
CHARLES BITTINGER, 3403 -O- Street, Washington, 7, D.C., U.S.A.
JAMES MARTIN BROWN, 88, Seacliffe Road, Bangor, N. Ireland.
PETER JOHN BUCKLEY, 38, Laing Street, Dunedin, S.W.1, New Zealand.
JACK M. CLARK, 3837, El Oro Street, North Highlands, Calif., U.S.A.
BELMONDO CLAUDIO, Corso Beccaria 2, Torino, Italy.
PETER AUSTIN COTTON, 20, Grange Park Terrace, Leeds, 8, Yorkshire.
FRANK LEWIS COX, 17, Ayloff's Walk, Hornchurch, Essex.
CYRIL CURTIS, The Cottage, Holme Lane, Bakewell, Derbyshire.
CHARLES THOMAS D'AIUTOLO, B.Ae.E., 125, South Fourth Street, Buckroe Beach, Virginia, U.S.A.
REGINALD JOHN DAVIS, 72a, Eversham Road, Redditch, Worcestershire.
CARLOS FEBLES, 3601, Parrish Avenue, East Chicago, Indiana, U.S.A.
ROBERT EARL FUGATE, 801, N. Washington Street, Brazil, Indiana, U.S.A.
WINSTON GIN, M.A., B.S., 11351, West Olympic Boulevard, Los Angeles 64, Calif., U.S.A.
GEOFFREY ERNEST GRIFFIN, "Woodlands," 48, Stoke Road, Linslade, Leighton Buzzard, Bedfordshire.
ALGIE WARNER HEARNE, 701, Bartlett Avenue, North Las Vegas, Nevada, U.S.A.
ROBERT HENRY HOLDHAM, 90, Channel View Road, Grange, Cardiff.
PETER HARRY KEMP, 26, The Glade, Baldock, Hertfordshire.
SIMON LOUIS KLEIN, 18415, Gottschalk Avenue, Homewood, Illinois, U.S.A.
ALFRED KNIGHT, 104, Highfield Road, South Shields, Durham.
KAZIMIERZ JOZEF KRZYZANOWSKI, Krakow 26, Czechowicza 13, Poland.
EUGENE LAWRENCE KUPSKI, B.S., 210, Crescent Avenue, Revere 51, Mass., U.S.A.
ADRIAN NICHOLAS MANDEVILLE, 24, Kidbrooke Park Road, Blackheath, London, S.E.3.
CHRISTIAN CHARLES MIESSE, M.A., 10 West 35th Street, Chicago 16, Illinois, U.S.A.
LEWIS DALE OWEN, 40, Rosshire, Pontiac, Michigan, U.S.A.
CHARLIE PIKOOS, 31, Pearse Street, Doornfontein, Johannesburg, Transvaal, South Africa.
LAWRENCE JOHN LECAN RAVENSCROFT, 157, Cavendish Road, South Chingford, London, E.4.
DERRICK RICHARDS, 65, Cliveden Avenue, Perry Barr, Birmingham, 22B.
MARY FOSTER ROMIG, B.A., 260, Palomar Street, La Jolla, Calif., U.S.A.

DUDLEY SKELLY, B.Sc., 1, Windeena Avenue, Pine Park, Johannesburg, South Africa.
DAVID PAUL STUHR, 20, Cleverdon Road, Ho-Ho-Kus, New Jersey, U.S.A.
ROBERT DAVID TURNER, 369, Seymour Street West, Kamloops, British Columbia, Canada.
WILLIAM WALLACE WEITHERS, Perry Gap, Roebuck Street, Bridgetown, Barbados, B.W.I.
ARCHIBALD CUTHILL WOOD, 17, Woodhouse Close, Perivale, Greenford, Middlesex.

The following elections were made at the Council Meeting on 15 June, 1957:

*Fellows.*

ROY E. BARTLE, M.S., 3209, Chicago Street, San Diego 17, Calif., U.S.A.
WILLIAM PARKES CARROLL, B.E.E., 8500, La Entrada, Whittier, Calif., U.S.A.
ALLAN B. CRUNDEN, M.A., M.D., 30, Porter Place, Montclair, New Jersey, U.S.A.
EDWARD LOVELL ELLIS, B.Sc., Grad.I.E.E., A.C.G.I., "Cranmere," School Lane, Gerrards Cross, Buckinghamshire.
HELMUT HOEPPNER, M.S., 1521, Leach Circle, Huntsville, Alabama, U.S.A.
B. SPENCER ISBELL, B.S., 509, Bluefield Avenue, Huntsville, Alabama, U.S.A.
RONALD NICHOLAS KELLY, Apt. No. 7, 33, Erlesmere Avenue, Brampton, Ontario, Canada.
PETER KERR, B.Sc., M.A., F.R.A.S., F.R.I.C., 22, Kingfield Road, Ealing, London, W.5.
Professor HENRYK MUSTER, M.Sc., Warszawa, ul. Mochnackiego 4m. 44, Poland.
Professor ZBIGNIEW PACZKOWSKI, Warszawa, ul. Mickiewicza 37m. 13, Poland.
GIULIO RICHARD PRIMAVESI, B.A., 220, Banstead Road, Banstead, Surrey.
MARY FOSTER ROMIG, B.A., 260, Palomar Street, La Jolla, Calif., U.S.A.
EMIL NEWTON ROWLAND, M.A., Ph.D., 45, Earls Court Road, Cardiff.
KOSTA SIVCEV, Dipl.Ing., Beograd, Senjacka 33, Yugoslavia.
THOMAS SZEKELY, B.S., M.E., 2217, Victory Parkway, Cincinnati 6, Ohio, U.S.A.
KAZIMIERZ ZARANKIEWICZ, Warszawa, ul. Wawelska 19 m. 20, Poland.

*Members.*

PERCY EDWARD ARGYLE, M.A., 4881, West Saanich Road, Royal Oak, British Columbia, Canada.
NORMAN ATKINSON, 40, Cornill Road, Southwick, Sunderland, County Durham.
CHARLES KEITH BUCHANAN, P.O. 217, Amsterdam, Ohio, U.S.A.
MICHAEL BUDDEN COX, 16, Archer's Road, Eastleigh, Hampshire.
MALCOLM CUTTER, 103, Musgrave Gardens, Durham City.
KENNETH HARRISON FORD, M.A., P.O. Box 386, Pacific Grove, Calif., U.S.A.
J. WILLIAM GRADY, B.S., Scientific and Technical Unit, U.S. Naval Forces, Germany, A.P.O. 757, U.S. Army.
MARK ANTONY RICHARD GUNSTON, B.Sc., c/o 178, Galleywood Road, Chelmsford, Essex.
JOHN ROBIN HIDDEN, 81, Beaconsfield Road, Blackheath, London, S.E.3.
RICHARD HIRST, "Four Trees," Hermitage Lane, East Grinstead, Sussex.
DAVID ROBERT LEWIS, 16a, Jealott's Hill, Bracknell, Berkshire.
RICHARD DEAN LINNELL, D.Sc., 1575, Rosegrans Street, San Diego 6, Calif., U.S.A.
ANTHONY ALFRED JOHN MARLEY, 36, Beechfield Road, Fremington, Barnstable, Devon.
DALE JOSEPH MORLAND, 239, Brandon Avenue, Brandon, Manitoba, Canada.
JOHN LIVINGSTONE MULLINS, 3, Heathery Tops, Scremerston, Berwick upon Tweed, Northumberland.
NICHOLAS CARTER NICHOLS, 129a, Upper Richmond Road, London, S.W.15.
IAN BURNIE PORTER, 97, St. Margaret's Avenue, Luton, Bedfordshire.
FRANK EDWARD REED, 89, Manor Square, Dagenham, Essex.
MICHAEL CHARLES STEVENS, 15, The Chine, Grange Park, London, N.21.
MARIE PHELPS SWEET, P.O. Box 96, Malibu, Calif., U.S.A.
GEORGE WINN, 244, Northumberland Park, Tottenham, London, N.17.
WILLIAM SYDNEY DURHAM WISE, B.Sc., Imperial College of Tropical Agriculture, Trinidad, B.W.I.

The following elections were made at the Council Meeting on 13 July, 1957:

*Fellows.*

PETER GRAHAM BOUD, 24, Cressy Road, Hampstead, London, N.W.3.
JOHN CHARLES DAVY, B.A., 22, Tudor Street, London, E.C.4.
ROSS FLEISIG, B.Ae.E., M.Sc., 58, Kilburn Road, Garden City, Long Island, New York, U.S.A.

JOHN SIDNEY GRIFFITH, B.Sc., 22, Vyne Road, Basingstoke, Hampshire.
JACK STANLEY GRIMSHAW, B.Sc., 321, Red Bank Road, Bispham, Blackpool, Lancashire.
SIDNEY JOHN HAMMOND, 2, Romney Road, Aylesbury, Buckinghamshire.
DANIEL THOMAS MEISENHEIMER, B.S., 50, Morris Lane, Milford, Connecticut, U.S.A.
ROBERT FREDERIC JENKIN MORGAN, 13, St. Matthews Road, Pontypool, Monmouthshire.
CHARLES WALTER MUNDAY, B.Sc., A.R.I.C., 40, Tiverton Drive, New Eltham, London, S.E.9.
GORDON WILLIAM PINHEY, B.Sc., 25, Sandhurst Road, Kingsbury, London, N.W.9.
DEREK WILLIAM SIDA, M.Sc., Ph.D., 42, Hollin Lane, Leeds, 16.
HARRY SINGER, B.Sc., Ph.D., A.M.I.Chem.E., 46, Clare Court, Judd Street, London, W.C.1.
ALEC JULIAN TYNDALE-BISCOE, Woodville House, Arbroath, Angus.

*Members.*

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OSWALD HORSLEY ROBINSON, B.A., Iwerne Stepleton, Blandford, Dorset.
HUGH TANSLEY, 3, The Mount, Wembley Park, Middlesex.
ALAN EDWARD WHEELER, 7, Rye Place, Crossgates, Leeds, 15.
BARRY WOODCOCK, 40, Inkerman Terrace, Leek, Staffordshire.
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The following elections were made at the Council Meeting on 17 August, 1957:

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HORACE EDWARD DALL, 166, Stockingstone Road, Luton, Bedfordshire.
GEOFFREY MICHAEL GARNER, B.Sc., 151, Stanbridge Road, Leighton Buzzard, Bedfordshire.
J. WILLIAM GRADY, B.S., Scientific and Technical Unit, U.S. Naval Forces, Germany. A.P.O. 757 U.S. Army, c/o Postmaster, New York, N.Y., U.S.A.
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MAIR SEARCH, B.Sc., The Old Mill, Moorlynch, nr. Bridgwater, Somerset.
DARROL STINTON, Palm Cottage, Well Street, Loose, Maidstone, Kent.
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# PERSONAL NOTES

PROFESSOR A. D. BAXTER (FELLOW), lately Deputy Principal and Professor of Aircraft Propulsion of the College of Aeronautics, Cranfield, has joined the de Havilland Engine Company, Ltd. as Chief Executive of the Rocket Engine Division. He took up his new appointment on 1 October, 1957, and will also take charge of the work of the de Havilland Engine Company's Nuclear Power Group.

# JOINT ACTIVITIES

**Ninth International Astronautical Congress**

The Eighth International Astronautical Congress held recently in Barcelona was a great success—a full report will appear in the next issue of *Spaceflight*.

In addition to the technical sessions, there were social functions such as visits to the monastery at Montserrat and to a champagne cellar, receptions and a banquet.

The 1958 Congress is to be held in the Netherlands during the last week in August, and members who are already thinking about next year's Summer holiday would do well to consider the possibility of combining their holiday with attendence at the Congress, which will probably be held in Amsterdam. The normal air fare is of the order of £12–14 (which compares very favourably with the cost for the more arduous sea trip) and this sum could be reduced if sufficient members attend to warrant chartering of a special aircraft by the Society. Congress fees (including the cost of preprints, banquet and excursion tickets) should not amount to more than £10, and hotel charges can be quite reasonable. The expenses for the week need not amount to more than £35–40 per person, even including a small budget for gifts and entertainments. In return one has the opportunity to meet leading astronautical personalities and to experience all the delights of a continental holiday, which can be quite a refreshing change from the Briton's usual visit to the seaside. Those who have more time and money to spare may like to extend their visit after the Congress is over (thus, it would be quite easy to visit the World's Fair, which is being held in Brussels next year from April to October).

It is of course rather too early to make definite plans, but this notice has been inserted so that members can consider attending the Congress before their holiday plans are too far advanced. This is an outstanding opportunity; subsequent Congresses held on the continent are likely to be organized in centres much more distant and are certain to be much more expensive. Members who are thinking of going to the Amsterdam Congress are asked to inform the Secretary of the probable number in their party, so that a preliminary assessment of the size of the British delegation can be obtained. Further announcements concerning the venue and programme of the Congress will appear in the *Journal*.

# REVIEWS

**Radio Telemetry.** By Myron H. Nichols and Lawrence L. Rauch. Second edition. 9 × 6 in. Pp. xiii + 461, illustrated. 1956. New York: John Wiley & Sons, Inc. ($12.00); London: Chapman & Hall, Ltd. (96s.).

In a rocket design project it is indisputable that radio telemetry must play one of the major roles in the successful and economic development of the vehicle. Since about 1930, applications for radio telemetry have multiplied at a considerable rate with the advent of pilotless or unmanned vehicles; a situation which can be concisely stated as the need to provide automatic, or synthetic, flight engineer-observers, as well as auto-pilots or control systems. In consequence of the rapid expansion in this field there has been a great outpouring of technical papers on systems, devices, applications and opinions. The authors have fortunately had the opportunity to view the field objectively, and to determine the basic underlying theories, and survey the current technique used in operational systems. They have eminently succeeded in their task and have produced a much needed fundamental text which has no counterpart in current literature.

The book is divided into three sections plus a set of appendices. The first section commences with a chapter which defines a telemetry system and includes a brief account of its historical development. The rest of this section comprises a further eleven chapters commencing with a study of the cause and effects of noise and errors which the authors describe as a "semi-qualitive presentation of the fundamentals with illustrations." This is somewhat of an understatement in that these chapters contain a considerable amount of analytical discussion, providing most of the important results, and also giving a good grounding in the terminology used. For economic reasons it is necessary to use a well-established technique in communication engineering, that of multiplexing. The two most important methods are fully discussed, namely, frequency-division multiplexing, and time-division multiplexing. This is followed by an equally clear study of several combined or hybrid variations of these two, which are found in modern systems. Chapter 4 contains a well organized discussion of the fundamentals of communication theory and all the various methods of modulation, including pulse modulation systems, which are fully dealt with. The next four chapters deal analytically with all the various multiplexing systems and the effects of cross-talk for each system. A further chapter studies the problem of "de-multiplexing," *i.e.*, interpreting the received data.

Chapter 11 deals at some length with comparisons between different multiplexing systems and some useful results are obtained. The final chapter in Part 1 considers very briefly the subject of pick-offs or instrumentation. This chapter never rises to the high standard of the rest of the book, which is a pity. However, the authors are aware of this and provide a good reference instead.

Part 2, comprising Chapters 13 and 14, returns again to the high standard of the rest of the book and provides a thorough mathematical treatment of the processes of modulation, multiplexing, and the concepts of frequency analysis.

Part 3, generally, contains more practical subjects and commences with a discussion of current U.S.A. equipments with liberal circuit and block diagrams to assist. A chapter is included on data reduction, and this time, on the unquestionable excuse of security, very little apart from the general concepts and needs is discussed. The final chapter on command link control of missiles using telemetry "in reverse" contains very little but common-sense, and has been included more as a talking point than as a seriously treated subject.

The Appendices provide theorems of various kinds, the results of which are required throughout the text, with the exception of Appendix 14, which is the U.S. Inter-Range Instrumentation Group Steering Committee Recommendations for Telemetry Standards (July 1955).

188 useful references are given and another good feature is a glossary of symbols, which are used consistently throughout.

There is no doubt that this excellently produced book, based on a shorter first edition which was privately published for the use of the authors' own students, fulfils a long felt need. It combines within its covers an assembly of all the basic theories required by the telemetry engineer. It does, however, fall short in that the input and output of the telemetry system is not so well catered for, due to the very much lower standard of the chapters on instrumentation and data reduction. Nevertheless, for those of us concerned with rocketry of one sort or another, this book, directly or indirectly, has put our future projects one step nearer fruition in that the authors have created a "science of telemetry" from a wide collection of hitherto, almost uncorrelated work in this field, and will get people thinking straight about the subject; an objective worthy of commendation.

K. C. GARNER.

**The Sun.** By Giorgio Abetti. Translated from the Italian by J. B. Sidgwick. $8\frac{1}{2} \times 5\frac{1}{2}$ in. Pp. 336, with 147 plates and 97 figures. 1957. London: Faber and Faber, Ltd. (63s.).

There is a well-known remark by Tolstoy's fictitious philosopher Prutkov to the effect that the Moon is more important than the Sun, because the Moon gives us light at night, when it is dark, while the Sun gives us light only in the daytime, when it is light anyway. We do not need a great knowledge of astronomy to see the fallacy of this argument, but the Sun does more than provide us with light. It keeps the surface temperature on Earth at a level which, through evolution, we have grown to consider as equable. It provides the source of nearly all the power that has been used on Earth—wind, hydroelectric, fossil fuel—until the coming of the nuclear age. To the meteorologist, it is the first cause of all the complicated atmospheric changes with which he has to deal. To the astrophysicist, it is, therefore, the best testing-ground of many astrophysical theories.

From early neolithic man, through the beginnings of astrophysics with Galileo's telescope, to modern rocket observations above the atmosphere, the Sun has been a favorite object of observation and study, both theoretical and practical. Indeed, our knowledge of the Sun today is most extensive; although much still remains to be done, we can give the lie to Shakespeare, who wrote of "the heaven's glorious sun, that will not be deep-searched by saucy looks; small have continual plodders won, save base authority from others' books."

There is no base authority in Professor Abetti's book. In spite of the very extensive literature of solar physics, the author has clearly covered critically a major part of it. Indeed, the first edition of this book, in 1934, very rapidly assumed the status of a standard work. This translation is of the second edition, of 1951. But solar physics moves quickly, and the translator and author have brought the book up-to-date to about 1954. It is the best, indeed perhaps the only, exhaustive modern treatment of the descriptive aspects of solar physics. Both the instruments and the observations made with them are admirably summarized, and illustrated with many fine photographs and diagrams. This book is of value, not only to the layman interested in astronomy, but also to the professional astronomer. True, the latter will have to find his theoretical discussions elsewhere, but as a summary of descriptive solar physics and solar-terrestrial relationships, I doubt if it could be bettered. And how pleasing it is to be able to say that the translation does justice to the original.

M. W. OVENDEN.


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